LOAN DOCUMENT

PHOTOGRAPH THIS SHEET

DTIC ACCESSION NUMBER

LEVEL

INVENTORY

0

WL-TR-97-4074

DOCUMENT IDENTIFICATION

Apr 97

DISTRIBUTION STATEMENT A

Approved for public release;
Distribution Unlimited

DISTRIBUTION STATEMENT

H A N D L E W I T H C A R E

ACCESSION FOR

NTIS GRA&I ☑
DTIC TRAC ☐
UNANNOUNCED ☐
JUSTIFICATION

BY

DISTRIBUTION/

AVAILABILITY CODES

DISTRIBUTION	AVAILABILITY AND/OR SPECIAL	
A-1		

DISTRIBUTION STAMP

DATE ACCESSIONED

DTIC QUALITY INSPECTED 1

DATE RETURNED

19970703 059

DATE RECEIVED IN DTIC

REGISTERED OR CERTIFIED NUMBER

PHOTOGRAPH THIS SHEET AND RETURN TO DTIC-FDAC

DTIC FORM JUN 90 70A

DOCUMENT PROCESSING SHEET

PREVIOUS EDITIONS MAY BE USED UNTIL STOCK IS EXHAUSTED.

LOAN DOCUMENT

WL-TR-97-4074

PROCEEDINGS OF THE ANNUAL MECHANICS OF COMPOSITES REVIEW (10$^{\text{TH}}$)

Sponsored by:

Air Force Wright Aeronautical Laboratories
Materials Laboratory

APRIL 1997

FINAL REPORT FOR PERIOD 15-17 OCTOBER 1984

Approved for public release; distribution unlimited

MATERIALS DIRECTORATE
WRIGHT LABORATORY
AIR FORCE MATERIEL COMMAND
WRIGHT-PATTERSON AFB OH 45433-7734

REPORT DOCUMENTATION PAGE			Form Approved OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave blank)*	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	April 1997	Final Report 15-17 OCT 1984

4. TITLE AND SUBTITLE

PROCEEDINGS OF THE ANNUAL MECHANICS OF COMPOSITES REVIEW (10th)

5. FUNDING NUMBERS

6. AUTHOR(S)

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Air Force Materials Laboratory
Nonmetallic Materials Division
Wright-Patterson AFB OH 45433

8. PERFORMING ORGANIZATION REPORT NUMBER

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

Materials Directorate
Wright Laboratory
Air Force Materiel Command
Wright-Patterson AFB Ohio 45433-7734
POC:Tammy Oaks, WL/MLBM, 937-255-3068

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

WL-TR-97-4074

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION AVAILABILITY STATEMENT

APPROVED FOR PUBLIC RELEASE; DISTRIBUTION IS UNLIMITED

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

This report contains the basic unedited vu-graphs of the presentations at the "Mechanics" of Composites Review" sponsored jointly by the Non-metallic Materials Division of the Air Force Materials Laboratory, the Structures Division of the Air Force Flight Dynamics Laboratory and the Directorate of Aerospace Sciences of the Air Force Office of Scientific Research. The presentations cover current in-house and contract programs under the sponsorship of these three organizations.

14. SUBJECT TERMS	15. NUMBER OF PAGES
epoxy-matrix composites; composite materials; resin matrix composites; composite bonded joints; fatigue of graphite/epoxy composites; fracture and fatigue of bi-materials	286
	16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
UNCLASSIFIED	UNCLASSIFIED	UNCLASSIFIED	SAR

AGENDA

MECHANICS OF COMPOSITES REVIEW

15-17 OCTOBER 1984

TUESDAY, 16 OCTOBER 1984 (CONTINUED)

WEDNESDAY, 17 OCTOBER 1984

FOREWORD

THIS REPORT CONTAINS THE ABSTRACTS AND VIEWGRAPHS OF THE PRESENTATIONS AT THE TENTH ANNUAL MECHANICS OF COMPOSITES REVIEW SPONSORED BY THE MATERIALS LABORATORY. EACH WAS PREPARED BY ITS PRESENTER AND IS PUBLISHED HERE UNEDITED. IN ADDITION, A LISTING OF BOTH THE IN-HOUSE AND CONTRACTUAL ACTIVITIES OF EACH PARTICIPATING ORGANIZATION IS INCLUDED.

THE MECHANICS OF COMPOSITES REVIEW IS DESIGNED TO PRESENT PROGRAMS COVERING ACTIVITIES THROUGHOUT THE UNITED STATES AIR FORCE, NAVY, AND NASA. PROGRAMS NOT COVERED IN THE PRESENT REVIEW ARE CANDIDATES FOR PRESENTATION AT FUTURE MECHANICS OF COMPOSITES REVIEWS. THE PRESENTATIONS COVER BOTH IN-HOUSE AND CONTRACT PROGRAMS UNDER THE SPONSORSHIP OF THE PARTICIPATING ORGANIZATIONS.

SINCE THIS IS A REVIEW OF ON-GOING PROGRAMS, MUCH OF THE INFORMATION IN THIS REPORT HAS NOT BEEN PUBLISHED AS YET AND IS SUBJECT TO CHANGE; BUT TIMELY DISSEMINATION OF THE RAPIDLY EXPANDING TECHNOLOGY OF ADVANCED COMPOSITES IS DEEMED HIGHLY DESIRABLE. WORKS IN THE AREA OF MECHANICS OF COMPOSITES HAVE LONG BEEN TYPIFIED BY DISCIPLINED APPROACHES. IT IS HOPED THAT SUCH A HIGH STANDARD OF RIGOR IS REFLECTED IN THE MAJORITY, IF NOT ALL, OF THE PRESENTATIONS IN THIS REPORT.

FEEDBACK AND OPEN CRITIQUE OF THE PRESENTATIONS AND THE REVIEW ITSELF ARE MOST WELCOME AS SUGGESTIONS AND RECOMMENDATIONS FROM ALL PARTICIPANTS WILL BE CONSIDERED IN THE PLANNING OF FUTURE REVIEWS.

F. D. Cherry

FRANKLIN D. CHERRY, CHIEF
NONMETALLIC MATERIALS DIVISION
MATERIALS LABORATORY

ACKNOWLEDGEMENT

WE EXPRESS OUR APPRECIATION TO THE AUTHORS FOR THEIR CONTRIBUTIONS AND TO THE POINTS OF CONTACT WITHIN THE ORGANIZATIONS FOR THEIR EFFORTS IN SUPPLYING THE PROGRAM LISTINGS.

A DAMAGE MODEL FOR CONTINUOUS FIBER COMPOSITES

by

D. H. Allen
S. E. Groves
Aerospace Engineering Department
and
R. A. Schapery
Civil Engineering Department
Texas A&M University
College Station, Texas 77843

It is now well known that ultimate failure of continuous fiber composite components is preceded by a sequence of microstructural events such as microvoid growth, transverse cracking, fiber-matrix debonding, interlaminar cracking, edge delamination, and fiber fracture which are all loosely termed damage. The significance of this damage lies in the fact that numerous global material properties such as stiffness and residual strength may be substantially altered during the life of the component.

A continuum mechanics approach is utilized herein to develop a model for predicting the thermomechanical constitution of continuous fiber composites subjected to both monotonic and cyclic fatigue loading. In this method damage is hypothesized to be characterized by a set of vector valued internal state variables representing locally averaged measures of matrix microvoids, transverse cracks, interlaminar delaminations, and fiber-matrix debonds. Utilization of thermodynamics with internal state variables [1] leads to constraints on the allowable form of constitutive relations. It is shown in the process that if the medium is initially elastic, the J-integral may be utilized to construct a quasi free energy function [2] which is given by

$$h_L = h_{E_L} - U_L^c,$$

where h_L is the locally averaged free energy, h_{E_L} is the locally averaged elastic free energy, and U_L^c is the locally averaged free energy due to cracking. The free energy for cracking is given by

$$U_L^c = U_L^c\,(\varepsilon_{L_{k\ell}}\ T_L,\ \alpha_i^j) \equiv \int_0^t G_{C_L}\ S'_2\ dt',$$

where $\varepsilon_{L_{k\ell}}$ is the strain tensor, T_L is the local temperature, α_i^j are the components of a set of vector valued internal state variables with j ranging from one to the number of damage modes [3,4], G_{C_L} is the local volume averaged energy release rate, $\dot{S}_2$ is the history dependent surface area of cracks due to damage, t is time of interest, and t′ is a dummy integration variable. The total free energy function is then expanded in a Taylor series expansion which leads to the following stress strain relations with damage:

$$\sigma_{L_{ij}} = \sigma_{L_{ij}}^R + C'_{L_{ijk\ell}}\ (\varepsilon_{L_{k\ell}} - \varepsilon_{L_{k\ell}}^T)$$

where $\sigma_{L_{ij}}$ and $\varepsilon_{L_{k\ell}}$ represent locally averaged measures of stress and strain, respectively, $\sigma_{L_{ij}}^R$ is the locally averaged damage dependent residual stress, $C'_{L_{ijk\ell}}$ is the damage dependent effective modulus

tensor, and $\varepsilon^T_{L_{k\ell}}$ is the locally averaged damage dependent thermal strain. The above relations are then simplified utilizing an irreducible integrity basis for transversely isotropic media. [5].

The resulting local constitutive relations are utilized in a laminate stiffness formulation in order to construct equations useful for experimental comparison. It is shown in this process that numerous experimentally determined parameters are required in order to characterize the model.

Finally, internal state variable growth laws are proposed for microvoids and transverse cracks and the model is compared to results obtain for $[0,90]_s$ and $[0,90_3]_s$ laminates with transverse cracks [6].

It is concluded that although the model requires further development and extensive experimental verification, it may be a useful tool in characterizing the constitutive behavior of continuous fiber composites with damage.

REFERENCES

1. Coleman, B.D., and Gurtin, M.E., "Thermodynamics with Internal State Variables," Journal of Chemical Physics, Vol. 47, pp. 597-613, 1967.
2. Schapery, R.A., "Models for Damage Growth and Fracture in Nonlinear Viscoelastic Particulate Composites," Proc. 9th U.S. National Cong. Appl. Mech., August, 1982.
3. Chou, P.C., Wang, A.S.D., and Miller, H., "Cumulative Damage Model for Advanced Composite Materials," AFWAL-TR-82-4-83, September 1982.
4. Wang, A.S.D., and Bucinell, R.B., "Cumulative Damage Model for Advanced Composite Materials," Interim Report No. 6, Feb. 1984.
5. Talreja, R., "A Continuum Mechanics Characterization of Damage in Composite Materials," The Technical University of Denmark, Lyngby, to be published in Proc. Royal Society, London.
6. Highsmith, A.L., Stinchcomb, W.W., and Reifsnider, K.L., "Stiffness Reduction Resulting from Transverse Cracking in Fiber-Reinforced Composite Laminates," Virginia Polytechnic Institute and State University, VPI-E-81.33, November, 1981.

A DAMAGE MODEL FOR CONTINUOUS FIBER COMPOSITES

by

D.H. Allen
S.E. Groves
R.A. Schapery

technical support by

W.L. Bradley
W.E. Haisler
C.E. Harris
V.K. Kinra
Y. Weitsman

Texas A&M University

financial support provided by

Air Force Office Of Scientific Research

OBJECTIVE

TO DEVELOP AN ACCURATE DAMAGE MODEL FOR PREDICTING STRENGTH AND STIFFNESS OF CONTINUOUS FIBER COMPOSITE MEDIA SUBJECTED TO FATIGUE OR MONOTONIC LOADING AND TO VERIFY THIS MODEL WITH EXPERIMENTAL RESULTS OBTAINED FROM COMPOSITE SPECIMENS OF SELECTED GEOMETRY AND MATERIAL MAKEUP.

PROCEDURE

- Utilize continuum mechanics with internal state variables (ISV)
- Hypothesize that damage is characterized by a set of vector-valued internal state variables
- Impose thermodynamic constraints
- Impose symmetry constraints
- Expand free energy in terms of ISV
- Obtain globally averaged (laminate)properties
- Construct ISV growth laws

KEY RESULTS TO DATE

- Free energy released during damage process is equivalent to fracture energy released
- Stress-strain relations completed for multiple damage modes (including residual & thermal stresses)
- Laminate equations completed for any stacking sequence with transverse cracking
- Laminate equations completed for $[0/90]_s$ with microcracking and transverse cracking
- Initial growth law constructed for transverse cracking
- Reduced stiffness FEM results obtained for laminates with transverse cracking
- Model comparisons to FEM (and some experimental) results for given damage surface areas due to transverse cracking
- Experimental observation of damage underway

HEIRARCHY OF CURRENT DAMAGE MODELS

DETERMINE ENERGY RELEASE RATES*

↓

PREDICT SURFACE AREA IN DAMAGED ZONES*

↓

PREDICT COMPONENTS OF DEGRADED STIFFNESS*

↓

PREDICT FAILURE

*UNDER STUDY IN THE TEXAS A&M MODEL

LANDMARK RESULTS WHICH INFLUENCE THE CURRENT MODEL DEVELOPMENT

- L.M. KACHANOV (1958)- FIRST DAMAGE PARAMETER
- COLEMAN & GURTIN (1965) - IMPOSITION OF THERMO-DYNAMIC CONSTRAINTS
- REIFSNIDER, ET AL. - CHARACTERISTIC DAMAGE STATE
- SCHAPERY (1981) - DAMAGE IN RANDOM PARTICULATE COMPOSITES
- A.S.D. WANG, ET AL. (1982) - ISV GROWTH LAW FOR TRANSVERSE CRACKING
- LAWS & DVORAK (1983) - SELF-CONSISTENT SCHEME FOR CRACKING
- TALREJA (1983) - SYMMETRY CONSTRAINTS ON REDUCED MODULUS TENSOR

EXPERIMENTAL OBSERVATION OF DAMAGE

- POST-CURE RESIDUAL DAMAGE
- MICROVOID GROWTH AND COALESCENCE*
- TRANSVERSE CRACKING
- INTERPLY DELAMINATION
- FIBER-MATRIX DEBOND
- EDGE DELAMINATION
- FIBER FRACTURE

*CURRENTLY UNDER INVESTIGATION IN THE MODEL

THERMODYNAMICS OF ELASTIC CONTINUA

CONSERVATION OF MOMENTUM

$$\sigma_{ji,j} = 0 \qquad \sigma_{ji} = \sigma_{ij}$$

CONSERVATION OF ENERGY

$$\rho\dot{u} - \sigma_{ij}\dot{\varepsilon}_{ij} + q_{j,j} = \rho r$$

SECOND LAW OF THERMODYNAMICS

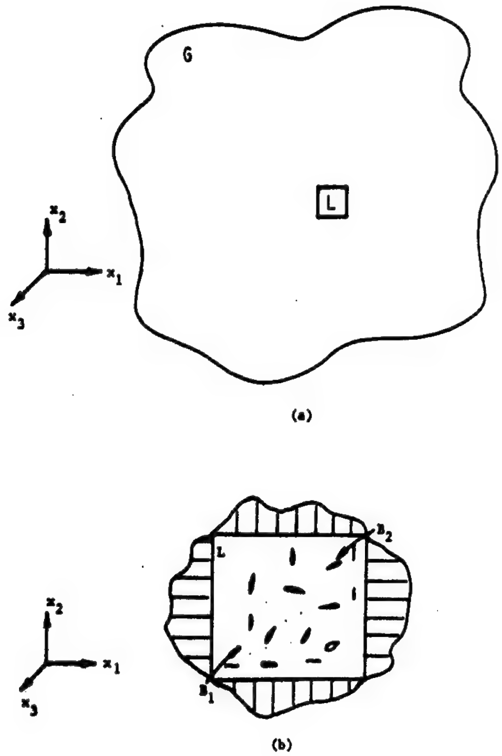
$$\rho\gamma = \rho\dot{s} - \frac{\rho r}{T} + \left(\frac{q_j}{T}\right)_{,j} \geq 0$$

STRAIN-DISPLACEMENT RELATIONS

$$\varepsilon_{ij} = 1/2\,(u_{i,j} + u_{j,i})$$

RESULTS

$$\sigma_{ij} = \sigma_{E\,ij} = \rho\,\frac{\partial h_E}{\partial \varepsilon_{ij}} = \rho\,(B_{ij} + C_{ijk\ell}\,\varepsilon_{k\ell} + E_{ij}\Delta T)$$



A Body with Damage (a) General Body, (b) Local Element.

THERMODYNAMIC CONSTRAINTS WITH LOCAL DAMAGE

DEFINITIONS

$$\sigma_{L_{ij}} \equiv \frac{1}{V_L}\int_{V_L} \sigma_{ij}\,dV \qquad \varepsilon_{L_{ij}} \equiv \frac{1}{V_L}\int_{B_1} u_i n_j\,ds$$

RESULTS

$$\sigma_L = \frac{\partial h_L}{\partial \varepsilon_{ij}}$$

WHERE

$$h_L \equiv h_{E_L} - u_L^C$$

$$\dot{u}_L^C \equiv \frac{1}{\rho_L V_L}\int_{B_2} \sigma_{ij}\dot{u}_i n_j\,ds$$

CONSTRUCTION OF A FREE ENERGY FUNCTION WITH DAMAGE

$$U_L^C = \int_{-\infty}^{t} \frac{1}{\rho_L V_L} G_{C_L} \dot{S}_2 \,dt$$

NOW ASSUME

$$U_L^C = U_L^C(\varepsilon_{L_{k\ell}}, T_L, \alpha_i^j)$$

DUE TO REFLECTIVE SYMMETRY

$$U_L^C = 0 \; |\alpha_i^j \alpha_k^\ell| \to$$

$$\alpha_i^j \alpha_k^l \propto \frac{\partial u_L^C}{\partial \varepsilon_{L_{ij}}} = \frac{\partial}{\partial \varepsilon_{L_{ij}}} \int_{-\infty}^{t} \frac{1}{\rho_L V_L} G_{C_L} \dot{S}_2 \,dt$$

EXPANDING h_L IN A T.S. EXPANSION GIVES

$$\sigma_{L_{ij}} = \sigma^R_{L_{ij}} + C'_{L_{ijk\ell}}(\varepsilon_{L_{k\ell}} - \varepsilon^T_{k\ell})$$

WHERE

$$\sigma^R_{L_{ij}} \equiv \rho_L (B_{L_{ij}} - L^{pq}_{k\ell ij}\,\alpha_k^p \alpha_\ell^q)$$

$$C'_{L_{ijk\ell}} \equiv \rho_L[C_{L_{ijk\ell}} - (P^{pq}_{mnk\ell ij} - T^{pq}_{mnk\ell ij}\Delta T)\alpha_m^p \alpha_n^q]$$

$$\varepsilon^T_{k\ell} \equiv C'_{L_{ijk\ell}}\, \rho_L(E_{L_{ij}} - Q^{pq}_{mnij}\,\alpha_m^p \alpha_n^q)\Delta T_L$$

$B_{L_{ij}}, L^{pq}_{k\ell ij}, C_{L_{ijk\ell}}, P^{pq}_{mnk\ell ij}, T^{pq}_{mnk\ell ij}, Q^{pq}_{mnij}$

= Material Constants

$\alpha_i^j \equiv$ vector valued internal state variables (ISV)

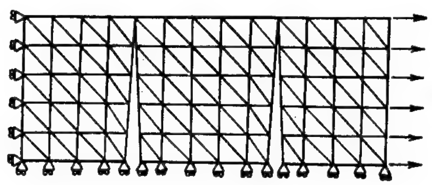
DETERMINE OF $P_{\alpha\beta}$

CONSTITUTIVE EQUATIONS

$$C'_{\alpha\beta} = C_{\alpha\beta} - \alpha^2 \; P_{\alpha\beta}$$

I. DETERMINE $C_{\alpha\beta}$ FROM EXPERIMENT ON UNDAMAGE MATERIAL

II. DETERMINE $C'_{\alpha\beta}$, G FROM FINITE ELEMENT ANALYSIS ON A $[90]_s$ LAMINA FOR TRANSVERSE CRACKING AS SHOWN



III. FOR GIVEN CRACK DENSITY DETERMINE α^2 AND THEN DETERMINE $P_{\alpha\beta}$

A FIRST GENERATION GROWTH LAW FOR TRANSVERSE CRACKING ($\bar{\alpha}^2$)

WANG, ET AL. (1982)

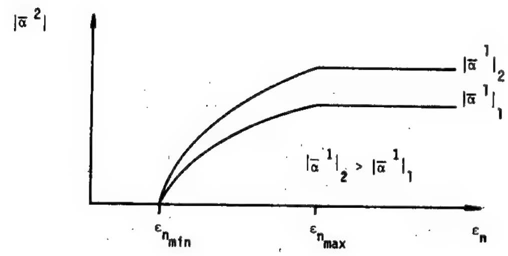
$$\frac{da}{dN} \; \alpha \; (G_c)^n$$

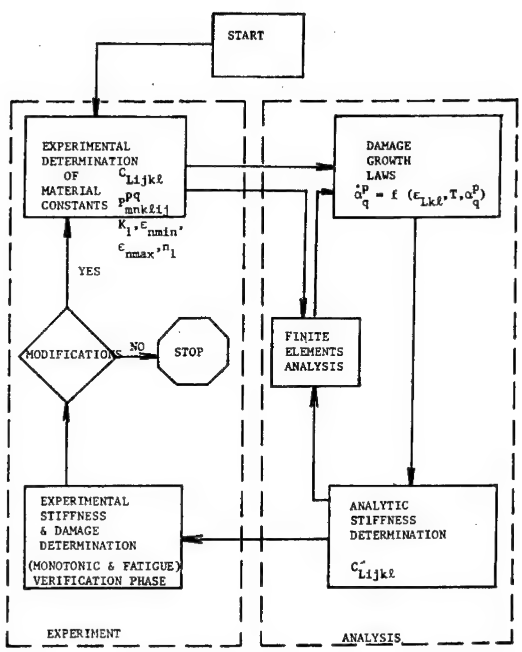
CURRENT MODEL

$$|\dot{\bar{\alpha}}^2| = k_1 \; |\dot{\bar{\alpha}}^1|^{n_1} \left(\frac{\epsilon_n - \epsilon_{n_{min}}}{\epsilon_{n_{max}} - \epsilon_{n_{min}}}\right)^{n_2} \frac{d\epsilon_n}{dt} \quad \text{if } 0 < \epsilon_{n_{min}} < \epsilon_n < \epsilon_{n_{max}}$$

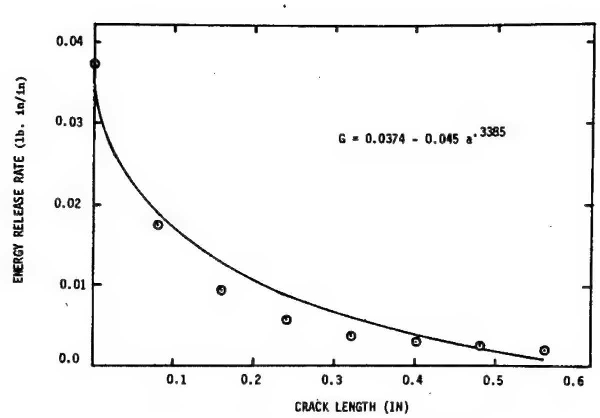
$$= 0 \qquad \text{if } \epsilon_n < \epsilon_{n_{min}} \text{ or } \epsilon_n > \epsilon_{n_{max}}$$

where $\bar{\alpha}^1$ = ISV FOR MICROVOIDS

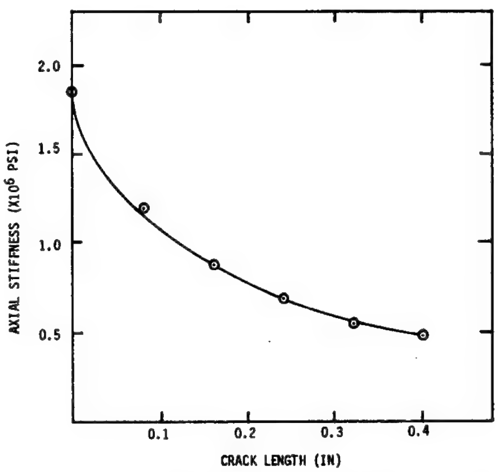




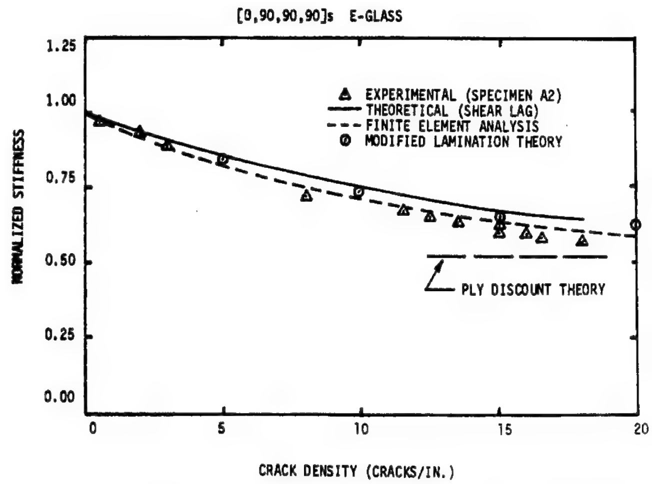
Flowchart for Model Development and Usage



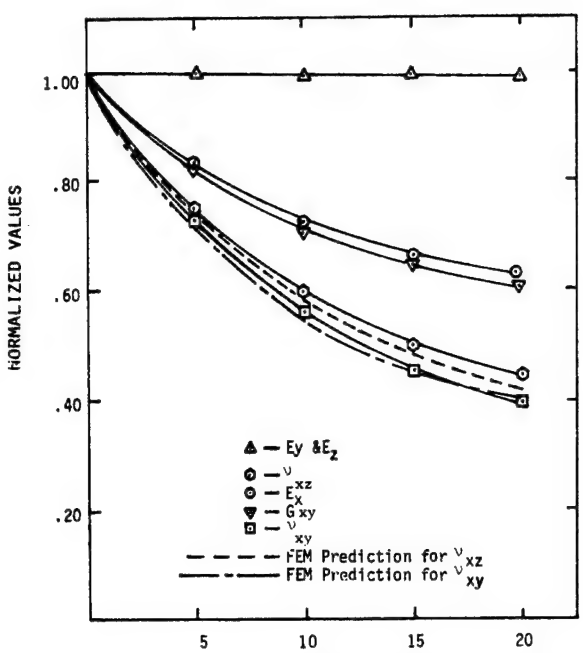
Energy Release Rate for Transverse Cracking in a $[90]_s$ lamina.



STIFFNESS REDUCTION IN $[90]_s$ LAMINATE FOR TRANSVERSE CRACKING

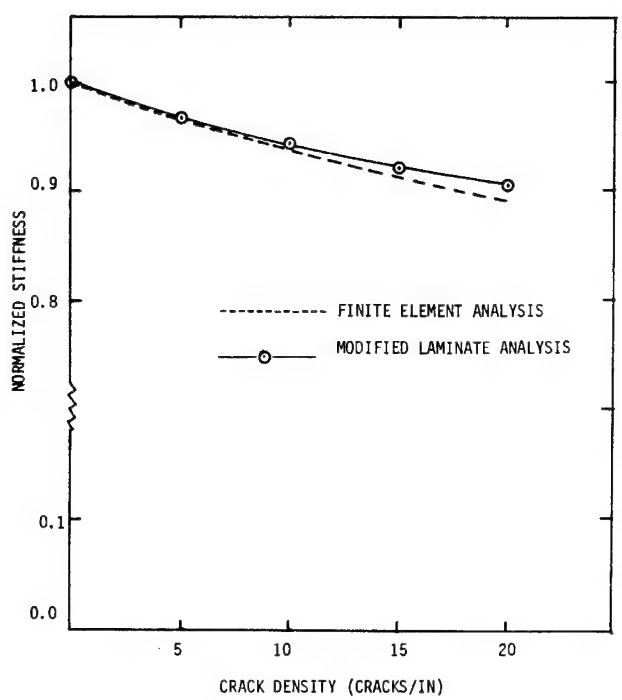


PREDICTIONS AND MEASUREMENTS OF STIFFNESS AS A FUNCTION OF CRACK DENSITY FOR A $[0,90_3]_s$ LAMINATE.

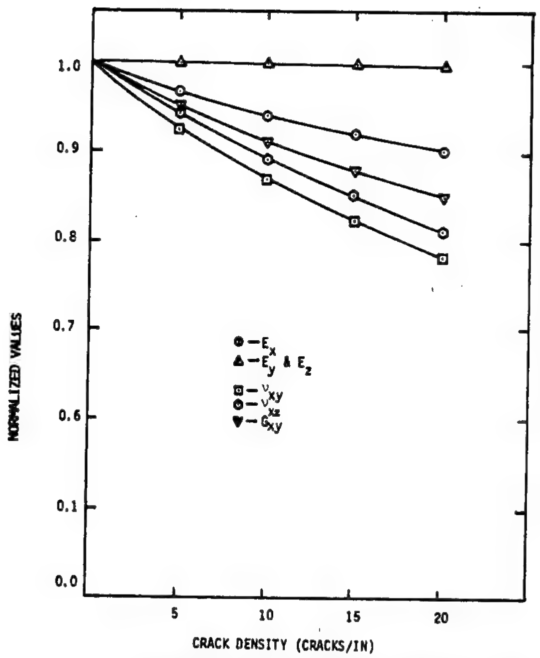


STIFFNESS REDUCTION FOR TRANSVERSE CRACKING IN A

$[0,90_3]_s$ LAMINATE.



STIFFNESS REDUCTION FOR TRANSVERSE CRACKING IN A
$[0,90]_s$ LAMINATE.



STIFFNESS REDUCTION FOR TRANSVERSE CRACKING IN A [0,90] LAMINATE USING MODIFIED LAMINATE ANALYSIS

CONCLUSION

- THE USE OF INTERNAL STATE VARIABLES IS NOW A WELL ESTABLISHED METHODOLOGY IN CONSTITUTIVE MODELING OF METALS AND RANDOM PARTICULATE COMPOSITE MATERIALS
- ONLY RECENTLY HAS THIS METHODOLOGY BEEN APPLIED TO CONTINUOUS FIBER COMPOSITES
- THE CURRENT RESEARCH INDICATES THAT ISV THEORY MAY BE APPROPRIATE FOR PREDICTING STIFFNESS REDUCTION DUE TO TRANSVERSE CRACKING
- FUTURE RESEARCH WILL DETERMINE WHETHER ISV THEORY IS APPROPRIATE FOR PREDICTING STIFFNESS REDUCTION AND LIFE WITH MULTIPLE DAMAGE MODES

GOALS FOR THE NEAR FUTURE

- IMPROVE ISV GROWTH LAW FOR TRANSVERSE CRACKING
- CONSTRUCT ISV GROWTH LAW FOR INTERLAMINAR DELAMINATION
- PERFECT NDE TECHNIQUES FOR MEASURING AND DISTINGUISHING DAMAGE MODES
- PERFORM COMPARISONS OF MODEL FOR VARIOUS LAYUPS TO EXPERIMENTAL AND FEM RESULTS

MATRIX CONTROLLED DEFORMATION AND

FRACTURE ANALYSIS OF FIBROUS COMPOSITES

R.A. Schapery
with Graduate Students
J.R. Weatherby
R.D. Tonda

Mechanics & Materials Center
Texas A&M University
College Station, TX 77843

ABSTRACT

Methods of quasi-static deformation and fracture analysis have been developed for nonlinear elastic, viscous, and viscoelastic materials with distributed damage [1]. The crack growth theory, which uses a generalized J integral that allows for viscoelasticity and distributed microscale damage, is not much more involved than that of nonlinear elasticity or special cases of linear viscoelasticity. This simplicity, compared to what one may expect, is a direct result of the particular type of constitutive equations and mechanical variables selected to characterize rheological behavior. Considering elastic materials with distributed damage, for example, the constitutive theory is expressed in terms of one strain energy-like potential for loading and another for unloading. The research activity is presently in the early stages of an investigation of the applicability of the theory to deformation and fracture of fiber-reinforced plastics. It is anticipated that additional information on its applicability will come from other AFOSR-sponsored projects at Texas A&M under the direction of Professors Allen, Bradley, Kinra, and Weitsman.

In this presentation we illustrate some features of the theory for elastic composite materials with damage and discuss current research activities. One important question is concerned with whether or not strain energy-like potentials actually exist; it has been addressed theoretically in [1] with encouraging results. Some examples of real nonlinear material behavior which can be characterized in this manner are given in this presentation, and then our experimental program to study this question for nonlinear behavior of unidirectional laminates is described. Another portion of the work is concerned with application of the theory to fracture characterization and analysis, assuming the requisite potentials exist. In this case the finite element method is being used to predict crack initiation and growth in materials with large-scale distributed damage, first for initially isotropic and homogeneous media and then for composites (i.e., delamination initiation and growth). A few years ago we employed linear elastic fracture mechanics in an investigation of the fracture behavior of a randomly oriented glass fiber reinforced plastic, SMC-R50 [2]. The data on this nonlinear material are reinterpreted here using J integral theory in order to further illustrate its use. Application of J integral theory to delamination growth when large-scale distributed damage exists is under study on Professor Bradley's project.

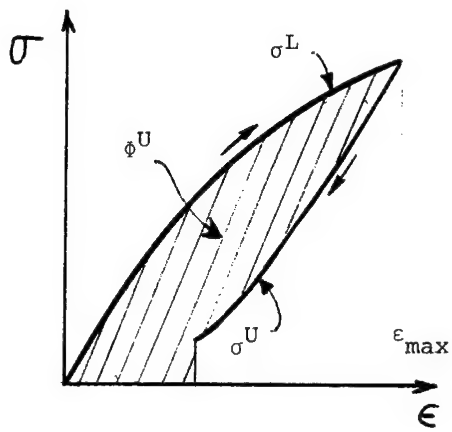
REFERENCES

1. R.A. Schapery, "Correspondence Principles and a Generalized J Integral for Large Deformation and Fracture Analysis of Viscoelastic Media," Int. J. Fracture, July 1984.

2. R.M. Alexander, R.A. Schapery, K.L. Jerina, and B.A. Sanders, "Fracture Characterization of a Random Fiber Composite Material," in Short Fiber Reinforced Composite Materials, ASTM STP 772, B.A. Sanders, Ed., pp. 208-224, 1982.

This research is sponsored by the Air Force Office of Scientific Research under grant AFOSR-84-0068.

CONSTITUTIVE THEORY USING POTENTIALS FOR MATERIALS WITH DAMAGE



Uniaxial Loading

$$\sigma^L = f(\varepsilon, \varepsilon) = \frac{d\phi^L}{d\varepsilon}$$

Unloading

$$\sigma^U = f(\varepsilon, \varepsilon_{max}) = \frac{\partial \phi^U}{\partial \varepsilon}$$

Multiaxial Loading

$$\sigma^L_{ij} = \partial \phi^L / \partial \varepsilon_{ij}$$

Unloading

$$\sigma^U_{ij} = \partial \phi^U / \partial \varepsilon_{ij}$$

FEATURES

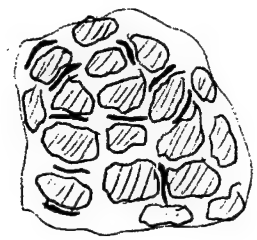
- CONVENIENT FORMULATION FOR FRACTURE APPLICATIONS
- DAMAGE PARAMETERS (E.G. ε_{max}) ARE EXPRESSED IN TERMS OF STRAIN HISTORY AND MAY BE RELATED TO PHYSICS OF DAMAGE PROCESS
- POTENTIALS ANALOGOUS TO STRAIN ENERGY DENSITY ARE USED FOR LOADING (ϕ^L) AND UNLOADING (ϕ^U)
- ALLOWS FOR TEMPERATURE AND MOISTURE INDUCED STRESSES
- VISCOELASTIC EFFECTS ARE INTRODUCED BY USING "PSEUDO DISPLACEMENTS" IN PLACE OF DISPLACEMENTS

APPLICATIONS

- EXISTENCE OF POTENTIALS SHOWN IN THE SPECIAL CASES OF PARTICLE-REINFORCED RUBBER, ELASTO-PLASTIC BEHAVIOR OF METALS AND SECONDARY AND TERTIARY CREEP OF METALS
- USE FOR FIBER-REINFORCED PLASTICS IS UNDER STUDY

CONSTITUTIVE EQUATION EXAMPLE

A SPECIAL PARTICULATE COMPOSITE WITH MICROCRACKING



$$\Phi^L = \int_0^W \left[1 - g(W')\right] dW'$$

$$\Phi^U = \left[1 - g(W_M)\right]\left[W - W_M\right] + \int_0^{W_M} \left[1 - g(W)\right] dW$$

where W = STRAIN ENERGY DENSITY WITHOUT DAMAGE ($g = 0$)

W_M = MAXIMUM W

- NOTE THAT $\Phi^L = \Phi^U$ WHEN $W = W_M$
- STRESSES ARE

$$\sigma_{ij}^L = \frac{\partial \Phi^L}{\partial \varepsilon_{ij}} = \left[1 - g(W)\right] \frac{\partial W}{\partial \varepsilon_{ij}}$$

$$\sigma_{ij}^U = \frac{\partial \Phi^U}{\partial \varepsilon_{ij}} = \left[1 - g(W_M)\right] \frac{\partial W}{\partial \varepsilon_{ij}}$$

GIVES REDUCED STIFFNESS
WHEN $0 < g \leq 1$

APPROACH FOR UNIDIRECTIONAL PLY CHARACTERIZATION

CONSIDER A UNI-AXIALLY LOADED, OFF-AXIS, UNIDIRECTIONAL COMPOSITE TENSILE SPECIMEN. THE STRESSES IN THE PRINCIPAL MATERIAL DIRECTIONS ARE WRITTEN IN TERMS OF THE STRAINS IN THOSE DIRECTIONS AS (NEGLECTING EFFECTS OF END CONSTRAINT):

$$\sigma_1 = \sigma_1(\varepsilon_1, \varepsilon_2, \gamma_{12}),$$

$$\sigma_2 = \sigma_2(\varepsilon_1, \varepsilon_2, \gamma_{12}),$$

$$\tau_{12} = \tau_{12}(\varepsilon_1, \varepsilon_2, \gamma_{12}).$$

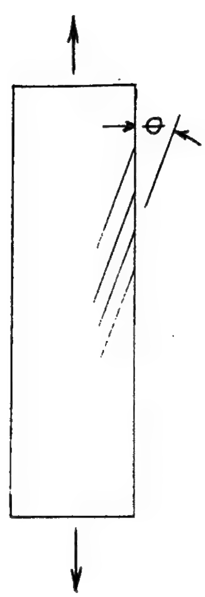
IF A POTENTIAL ϕ IS TO EXIST, SUCH THAT

$$\sigma_1 = \partial\phi/\partial\varepsilon_1, \quad \sigma_2 = \partial\phi/\partial\varepsilon_2, \quad \tau_{12} = \partial\phi/\partial\gamma_{12}$$

IT IS NECESSARY THAT, FOR EXAMPLE,

$$\left.\frac{\partial\sigma_2}{\partial\gamma_{12}}\right|_{\varepsilon_1, \varepsilon_2 = \text{CONST.}} = \left.\frac{\partial\tau_{12}}{\partial\varepsilon_2}\right|_{\varepsilon_1, \gamma_{12} = \text{CONST.}}$$

- PERFORM TESTS ON AS4/3502 MATERIAL USING OFF-AXIS SPECIMENS.



Θ = FIBER ANGLE

BY VARYING Θ, THE NECESSARY DATA CAN BE DEVELOPED TO EVALUATE THE CROSS-DERIVATIVES AND DETERMINE THE EXISTENCE OF ϕ.

THESE DATA ALSO CAN BE USED TO DEVELOP THE POTENTIAL ϕ ITSELF.

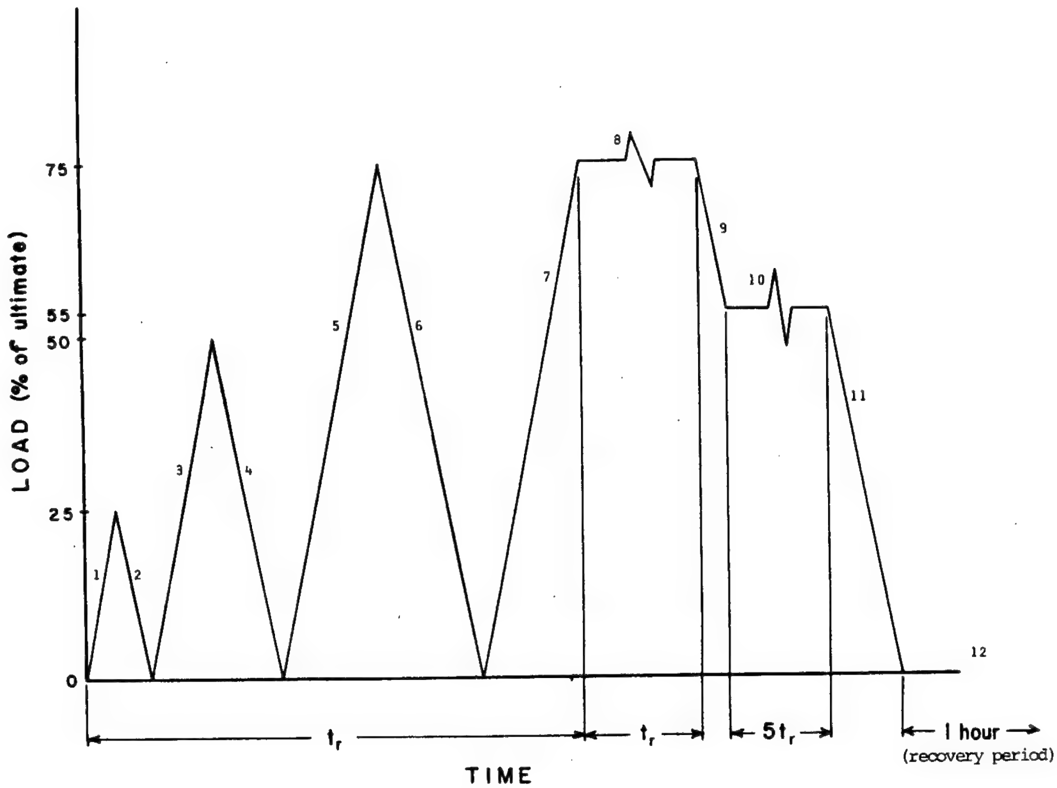


Figure 1. Load-time history used in 30° off-axis tests. (tr = 210 sec.)

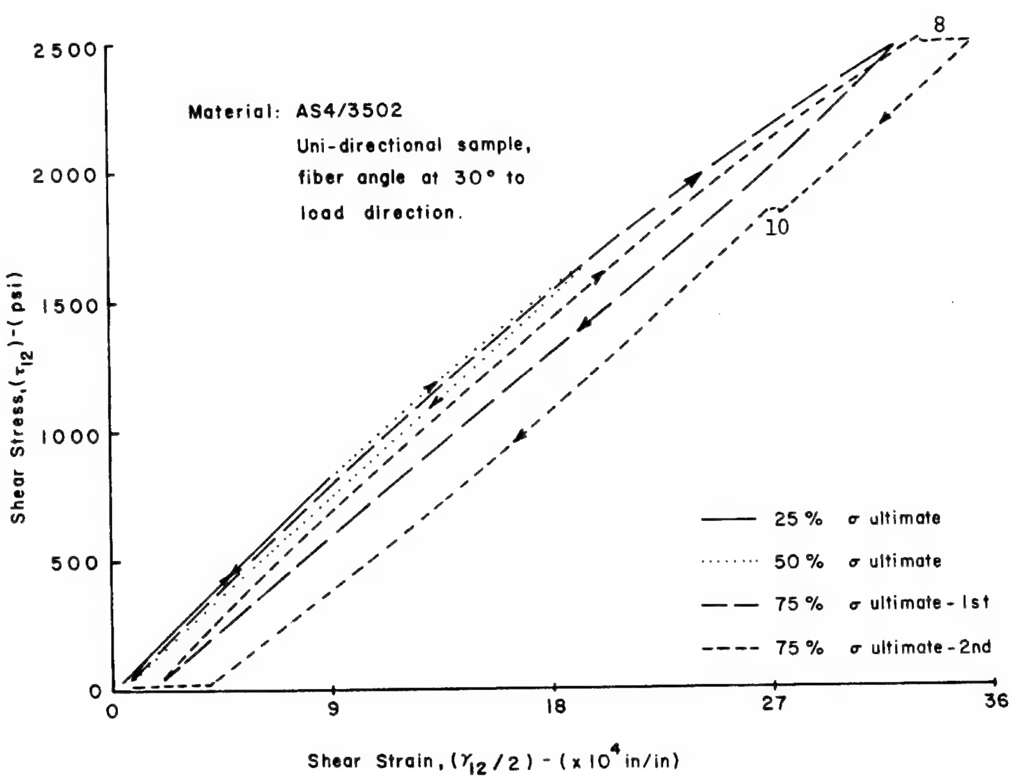


Figure 2. Shear stress versus shear strain from test in Figure 1.

FRACTURE EXAMPLE

- FRACTURE OF EDGE-NOTCHED COUPONS OF A SHORT FIBER COMPOSITE, SMC-R50

LOAD-DISPLACEMENT RELATION:

$$P = Cu^n , \quad n = 0.78$$

WHERE

$$C = C(a) , \quad a = \text{depth of edge notch}$$

LOADING POTENTIAL ("STRAIN ENERGY"):

$$\Phi = \int_0^u P du = Cu^{(n+1)}/(n+1)$$

J INTEGRAL ("ENERGY RELEASE RATE"):

$$J = -\frac{1}{B}\frac{\partial \Phi}{\partial a} = -\frac{u^{n+1}}{(n+1)B}\frac{dC}{da}$$

SET $J = J_c$ AND SOLVE FOR FRACTURE DISPLACEMENT u_f AND THEN FRACTURE STRESS σ_f.

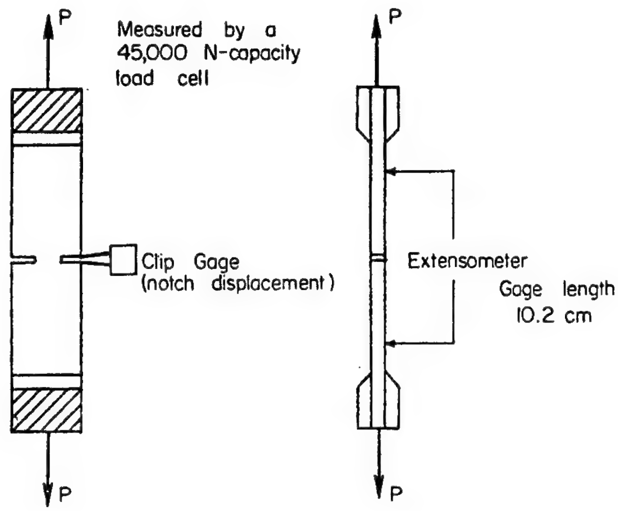


Figure 3. Double edge notch specimen with measured parameters indicated. (SMC-R50)

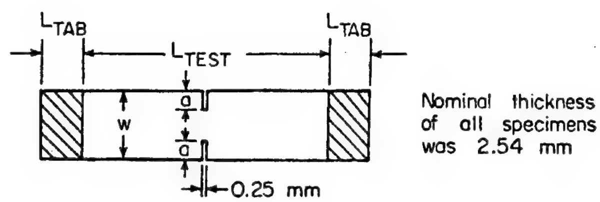


Figure 4. Double edge notch specimen geometry. (SMC-R50)

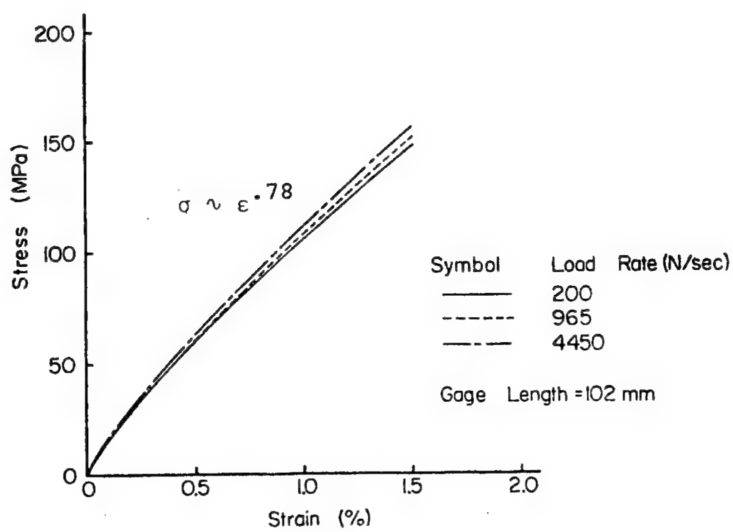


Figure 5. Stress-strain curves based on load-displacement data for 25.4 mm-wide unnotched tensile specimens. (SMC-R50)

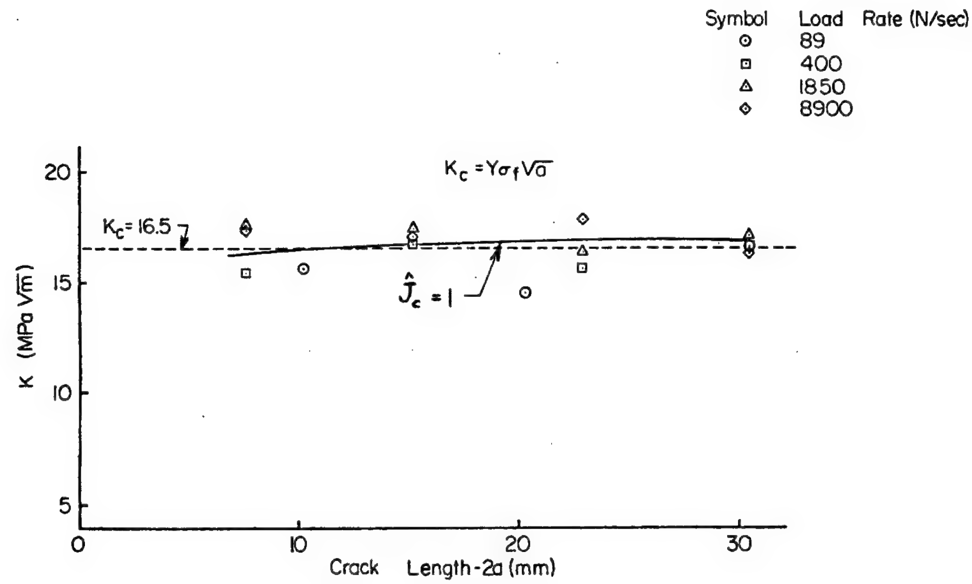


Figure 6. Critical stress intensity factor for the 50.8 mm-wide specimens. (SMC-R50)

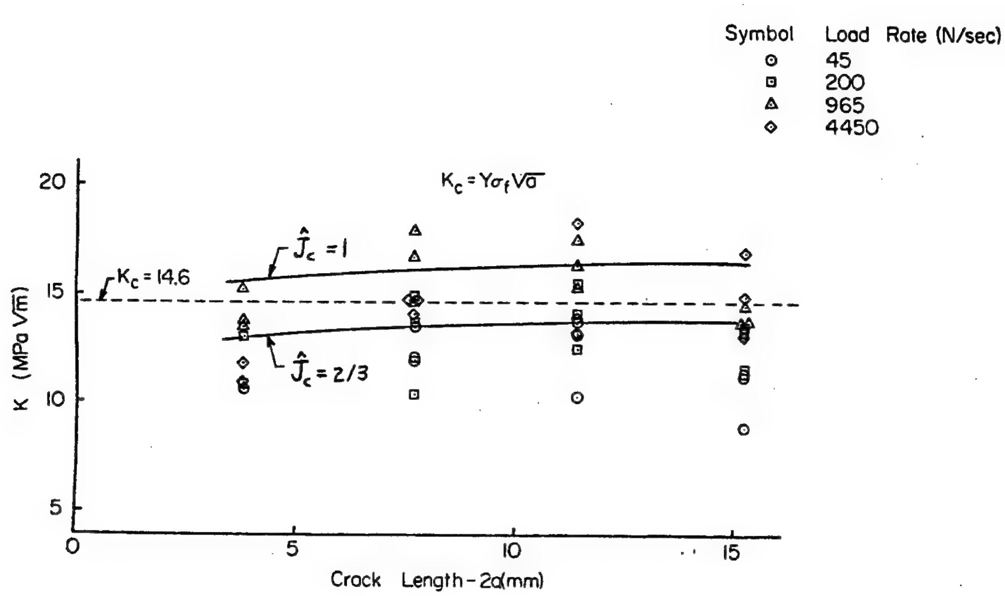


Figure 7. Critical stress intensity factor for the 25.4 mm-wide specimens. (SMC-R50)

ANALYSIS OF CRACK GROWTH IN DAMAGED MEDIA USING A GENERALIZED J INTEGRAL

- ANALYZE THE INITIATION AND PROPAGATION OF A CRACK IN A MATERIAL WITH DAMAGE THROUGH THE USE OF A FAILURE ZONE CRACK TIP MODEL AND THE GENERALIZED J INTEGRAL.

INITIALLY, THE ISOTROPIC MATERIAL MODEL OF J_2 DEFORMATION THEORY OF PLASTICITY, MODIFIED BY ELASTIC UNLOADING, IS BEING USED:

$$\sigma'_{ij} = 2G_d(\tau_s)\varepsilon'_{ij}\,, \qquad \text{for} \quad \tau_s = (\tau_s)_{max}$$

$$\sigma'_{ij} = \sigma'^{m}_{ij} = 2G(\varepsilon'_{ij} - \varepsilon'^{m}_{ij})\,, \qquad \text{for} \quad \tau_s < (\tau_s)_{max}$$

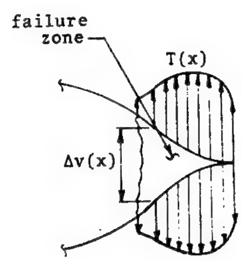
$$\sigma_{kk} = 3K\varepsilon_{kk}$$

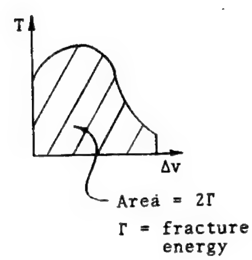
WHERE: $$\sigma'_{kj} = \sigma_{ij} - \frac{1}{3}\delta_{ij}\sigma_{kk}$$

$$\tau_s = \left(\frac{1}{2}\sigma'_{ij}\sigma'_{ij}\right)^{1/2}$$

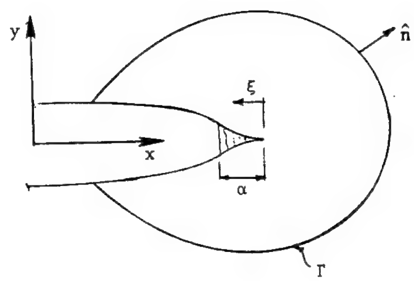
EXTENSION TO ORTHOTROPIC COMPOSITES WITH DAMAGE TO FOLLOW

- FAILURE ZONE MODEL FOR THE CRACK TIP:





- THE J INTEGRAL IS USED TO OBTAIN INFORMATION ABOUT WORK INPUT TO THE FAILURE ZONE FROM REMOTE FIELD QUANTITIES:



$$J = \int_{\Gamma} (\Phi n_1 - T_i \frac{\partial u_i}{\partial x})\, ds$$

$$J_f = \int_0^{\alpha} T_i \frac{\partial \Delta u_i}{\partial \xi}\, d\xi$$

$$J = J_f$$

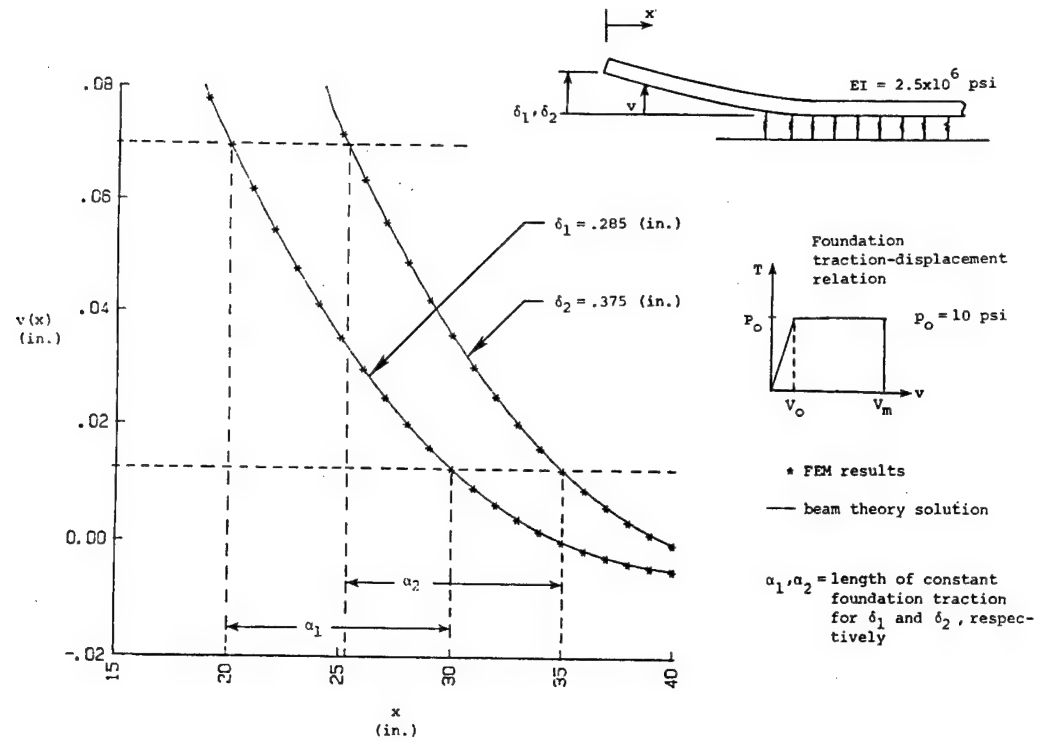


Figure 8. Deflections in a cantilever supported by an elastic-plastic foundation in which the tractions vanish above a critical displacement.

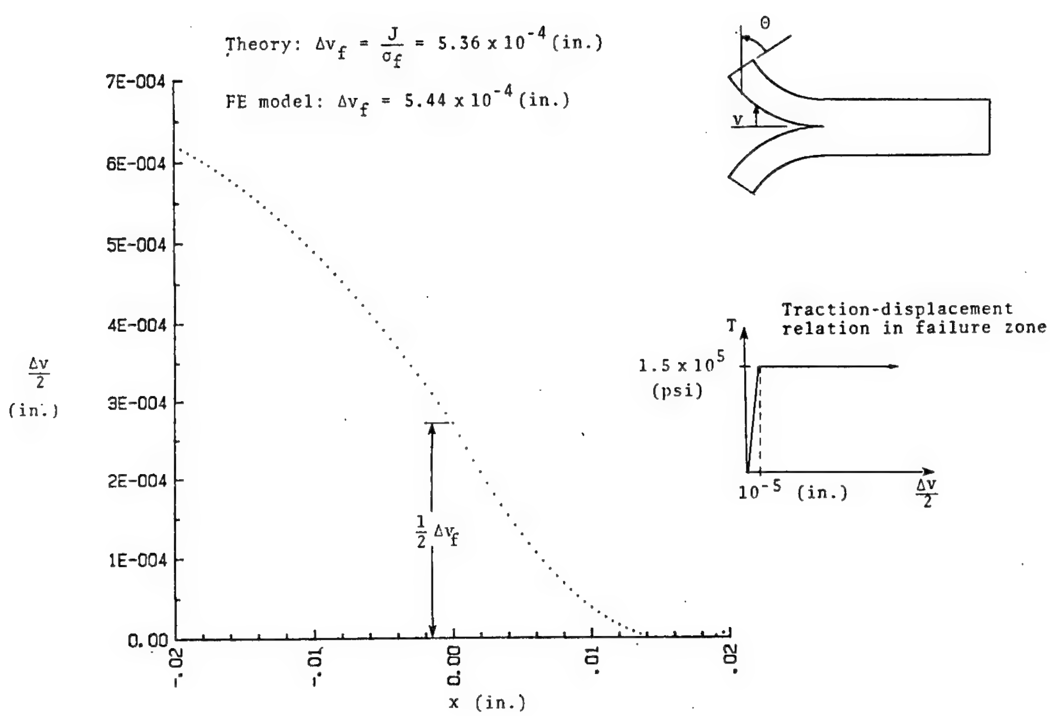


Figure 9. Crack opening displacement profile for thick, nonlinear bend specimen.

A CONSISTENT SHEAR LAG THEORY FOR FIBER-REINFORCED COMPOSITE MATERIALS

G.P. Sendeckyj and W.F. Jones

Flight Dynamics Laboratory
Air Force Wright Aeronautical Laboratories
Wright-Patterson AFB, OH 45433

ABSTRACT

A large body of experimental data, accumulated over the past twenty years, suggests that the occurrance of a small number of breaks in contiguous fibers is the critical event in the failure process in fiber-reinforced, resin-matrix composite materials. The breaks in the contiguous fibers can be thought of as forming a small "crack" that controls failure. The critical "crack" size (number of broken contiguous broken fibers) has been observed to be insensitive to the presence of matrix cracking and delaminations. The matrix damage only affects the load under which the "crack" propagates unstably. Based on these observations, a number of statistical theories have been formulated for predicting the longitudinal strength of unidirectional composite materials. Extension of these statistical strength theories to laminates is straightforward, requiring the evaluation of stress concentration factors for fibers near broken fibers in the presence of matrix damage. The required stress concentration factors can be determined by using shear lag theories. Unfortunately, the available shear lag theories are either based on overly simplifying assumptions or not consistent with the theory of elasticity.

In the present paper, a procedure is presented for deriving shear lag theories of any desired order of accuracy. The derivation procedure consists of modeling the unidierectional composite as a continuum, using the method of lines to replace the field equations by a system of difference-differential equations, and interpreting the material constants in the difference-differential equilibrium equations by using micromechanics. This procedure yields shear lag theories that are consistent with elasticity theory and account for transverse normal stresses and the Poisson effect.

The derivation procedure is illustrated by working out the first and second order shear lag theories for a two-dimensional model of the composite. The resulting difference-differential equations are reduced to a system of ordinary differential equations that have essentially identical structure that does not depend on the order of the shear lag theory. The computations involved in working out specific examples are straightforward, requiring the numerical evaluation of a small number of definite integrals.

The application of the new shear lag theories is illustrated by determining the stress concentrations in fibers that are adjacent to broken ones. The results show that the second order theory is superior to the first order theory. The first order theory does not exhibit the oscillations and cusps encountered in the Eringen and Kim shear lag theory and reduces to the Hedgepeth theory upon setting the transverse Young's modulus and Poisson's ratio equal to zero. Moreover, both new shear lag theories give the "correct" results for the transverse matrix stresses.

DELAMINATION FRACTURE OF GRAPHITE/EPOXY COMPOSITE MATERIALS

by

Walter L. Bradley and William Jordan
Texas A&M University
College Station, Texas 77843

The objectives of this research program have been (1) to better define the deformation and fracture physics of delamination fracture in graphite/epoxy composite materials so that realistic micromechanics models of matrix dominated fracture can be developed and (2) to develop and refine reliable experimental and analytical techniques to measure mode I, mode II and mixed mode delamination fracture toughness G_{1c} of both unidirectional and multidirectional composite laminates to provide a benchmarks against which predictions of the various micromechanics models may be tested as well as provide reliable design data.

The experimental approach used in this program to measure delamination fracture toughness G_c in symmetrically or symmetrically loaded split laminates, is shown in Figure 1. Depending on the degree of asymmetry, one can obtain any combination of mode I/ mode II loading desired, including pure mode II. When doing pure mode II loading as shown in 1A, a small spacer is used between the split laminate arms to avoid additional resistance to crack growth due to friction. Subsequent examination of the specimens in the scanning electron microscope (Figure 7) gave no evidence of rubbing between the two arms.

Results of our work for pure mode I and pure mode II delamination tests on AS4/3502 are presented in Table I and compared with results on a similar material, AS1/3501-6, by several other investigators using different experimental approaches. Table II contains additions results for G_c for intermediate values of mode II/mode I for AS4/3502 and Hexcel T6T145/F155. Our higher mode I G_{1c} values for the AS4/3502 when compared to mode I G_{1c} results reported by others on the similar AS1/3501-6 are due to a greater degree of fiber bridging in the AS4/3502 system. The ratio of pure mode II G_{IIc} to pure mode I G_{Ic} for our work is intermediate to that reported by the other investigators.

The microscopic observation of the fracture processes as they occur is made possible by delaminating specimens in the scanning electron microscope and recording the observation on video tape or film. A careful polishing of the surface to be observed prior to delamination of the specimens in the SEM allows one to observe the crack path through the resin or fiber/resin interface and the microcracking in the matrix that sometimes accompanies the fracture process. A panel of bled T6T145/F155 and a panel of unbled T6T145/F155 (manufacturer recommends unbled) were tested macroscopically as split laminate and also were fractured in the SEM. The unbled specimen with the thicker resin rich region between plies had a G_{1c} of approximately $1000 J/m^2$ whereas the bled specimen had a much thinner resin rich region between plies and a G_{Ic} of only $600 J/m^2$. In-situ fracture results from the SEM are seen in Figure 2A-2E. The unbled material had a very high density of microcracking in the resin rich region between plies with crack advance occurring sometimes by coalescence of these microcracks and sometimes by debonding at the resin/fiber interface. The bled material by contrast showed a much lower density of microcracking, but a greater volume of material so damaged, including resin several fiber diameter removed from the primary crack plane. The microcracking in the unbled specimen, by contrast, was contained almost exclusively in the somewhat thicker resin rich region between plies.

In-situ delamination of AS4/3502 in the SEM for pure mode I loading is seen in Figure 2D, which should be contrasted with similar results for T6C145/F155 shown in Figure 2A-2C. No microcracking is seen

and no evidence of microcracking such as hackles is seen during a post-mortem examination of the fracture surface. While the resin is quite brittle, cracking still occurs most often along the fiber/resin interface in this system. These observations seem consistent with the much lower G_{Ic} measure for this system compared to the T6C145/F155.

Fracture of the neat F155 resin in the SEM is also seen to occur by microcracking around the crack tip followed by coalescence of microcracks. The microcracking was very similar to that observed in the delamination fracture of the composite, though the extent was much greater, being approximately 3X the distance above and below the primary crack plane and 2X as far in advance of the crack tip (see Figure 3A). Fractographic examination of the surface after the specimen is fractured revealed a very faceted surface which again suggest failure primarily by coalescence of microcracks (see Figure 3B) Thus, the toughness of the F155 resins seems to be due to a significant incidence of microcracking rather than to a significant amount of flow. However, observation of the faceted fracture surface of the F155 at a higher magnification does show some local flow (see Figure 3C).

Increasing the fraction of mode II loading gives an increasing total energy release rate and an increased incidence of hackles or scallops on the fractured surface, as seen in Figures 4-7. A careful examination of the hackles indicates that cracking in this brittle system under increasing mode II loading gives a increasing angle for the microcracking ahead of the crack tip with the cracks assuming a sigmodial shape to allow for coalescence. The correlation between degree of mode II loading and hackle orientation suggests in this brittle system that the fracture mechanism is brittle fracture on the principal normal stress plane. These short microcracks cannot generally propagate beyond the graphite fibers. Thus, cracking proceeds in the plane between plies by coalescence of these microcracks, which gives the sigmodial shape.

Finally, the strain in the region surrounding the crack tip has been measured using a technique originally developed by Davidson and Lankford of Southwest Research Institute for use on metallic specimens. The crack tip region is photographed in the SEM in the unloaded condition and again in the loaded condition. Image analysis is used to measure the relative displacements between various artifacts on the surface of the two photographs, from which the strain at various points can be calculated. This technique, thus, allows the strain field around the crack tip to be quantified, as seen in Figure 8 for a single edge notched unidirectional composite loaded perpendicular to the fiber direction.

TABLE I. A COMPARISON OF G_c VALUES OBTAINED BY DIFFERENT METHODS

Source	Material	Types of Tests	G_{Ic} (J/m^2)	G_{tot} (76% mode II)	G_{II}
Ref.1	AS1/3501-6	DCB,CLS,3PT BEND	132	325	458
Ref.2	AS1/3501-6	Arcan	80		800
Ref.3	AS1/3501-6	DCB,CLS	130	460	---
PRESENT WORK	AS4/3502	DCB,ASYM. DCB	190		1600

TABLE II. G_{Ic} RESULTS FOR VARYING FRACTION OF MODE II LOADING

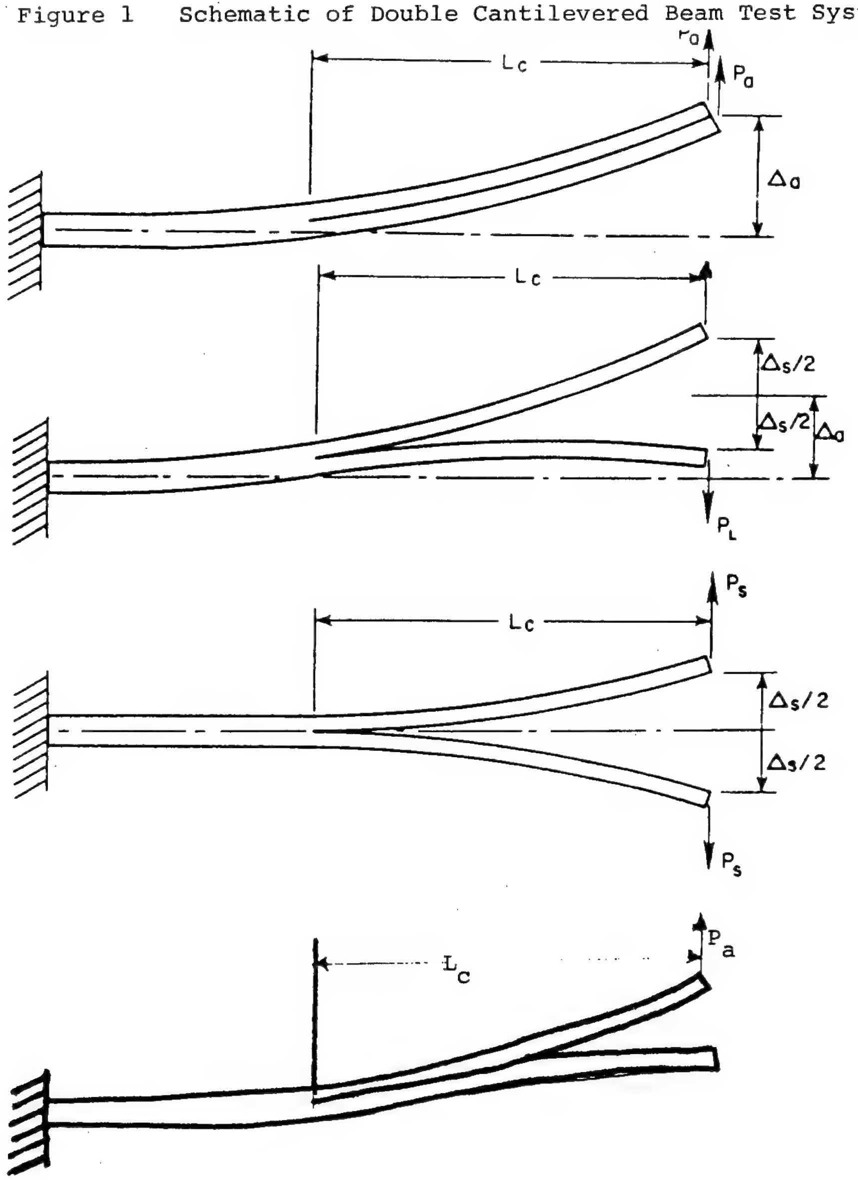
AS4/3502 % MODE II	AS4/3502 G_{Ic} (J/m^2)	T6T145/F155 % MODE II	T6T145/F155 G_{Ic} (J/m^2)
0	190	0	431
10	160	11	530
17	188	21	535
		31	485
43	765	43	481
100	1600	100	2630

[1] A.J. Russell and K.N. Street, "Moisture and Temperature Effects on the Mixed-Mode Delamination Fracture of Unidirectional Graphite/Epoxy", presented the ASTM Symposium on "Delamination and Debonding", 1983.

[2] Adapted by Russell and Street from K_c values in "Interlaminar Fracture of Composite Materials by Robert A. Jurf and R. Byron Pipes, J. of Composite Materials, Vol. 16, pp.386-394.

[3] D.J. Wilkins, "A Comparison of the Delamination and Environmental Resistance of a Graphite-Epoxy and a Graphite-Bismaleimide", NAV-GD-0037, General Dynamics, Fort Worth, Texas, September, 1981.

Figure 1 Schematic of Double Cantilevered Beam Test System



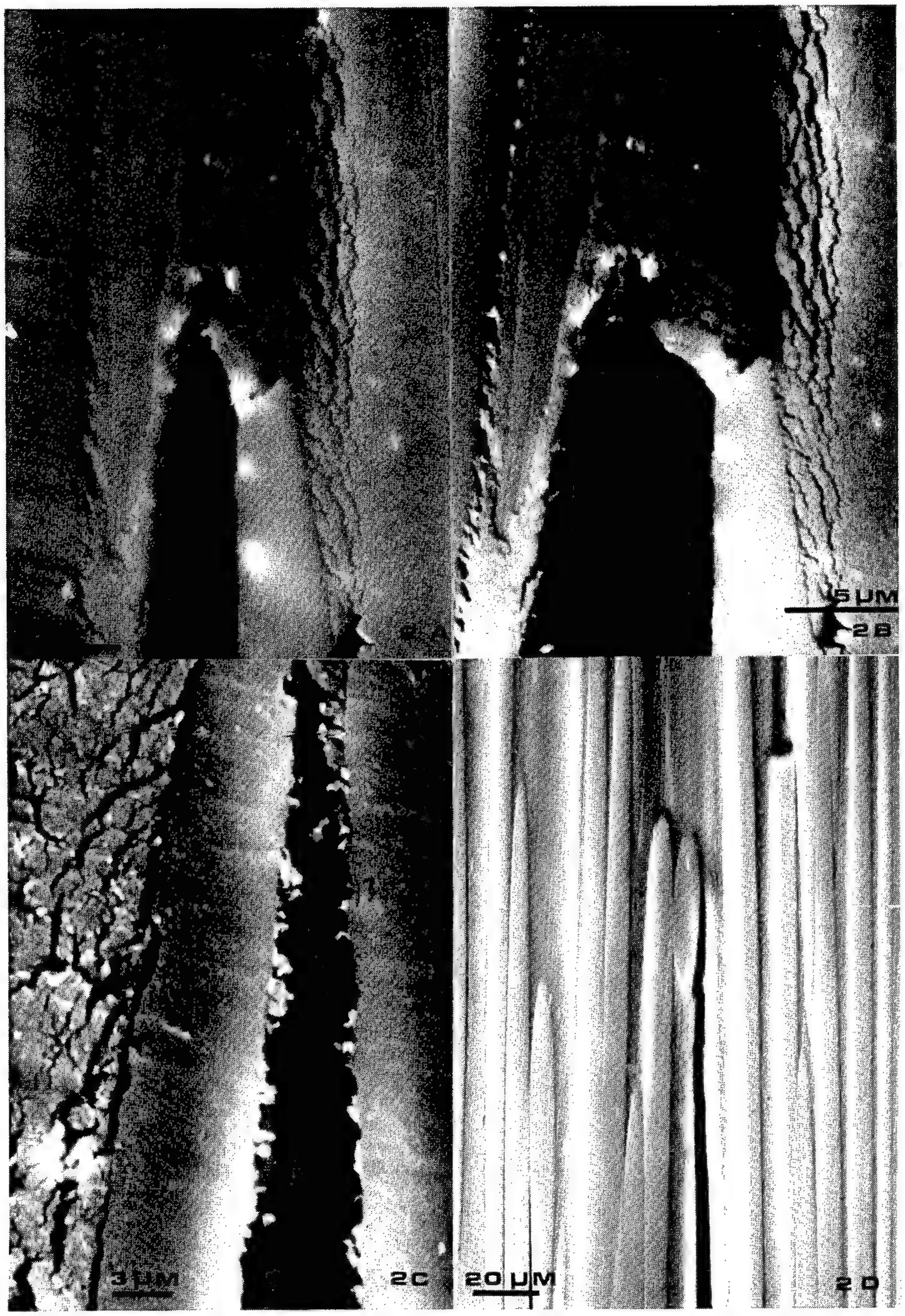
1A Asymmetric Loading -Mode 2

1B Mixed mode load Combination of 1A (mode 2) and 1C (mode 1)

1C Symmetric Load -Mode 1

1D Special case of 1B where $P_L=0$ This results in a sample that is in 43% Mode 2 the entire test.

This test apparatus provides a way to measure G_c for pure Mode 1 delamination by applying a symmetric load (1C). A varying percentage of Mode 2 throughout the test can be obtained by the asymmetric loading shown in 1B. A test that is a constant 43% Mode 2 throughout the test can be done by using the special case of 1D. Finally pure Mode 2 results can be obtained by applying loading as shown in 1A.

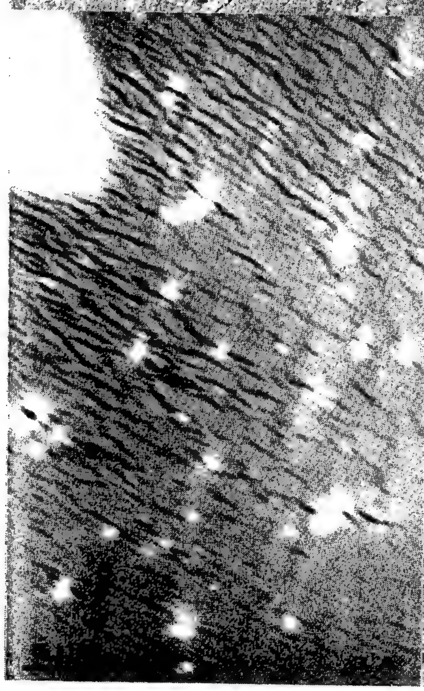


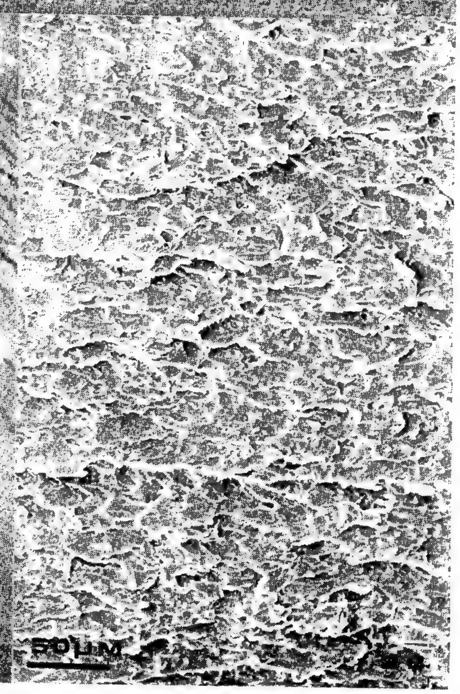
2A In-situ delamination of F155 composite (unbled during cure) (3900 x)

2B In-situ delamination of F155 composite(material in 2A after some additional crack growth) (3900 x)

2C In-situ delamination of F155 composite (bled during cure). (3400 x)

2D In-situ delamination of AS4/3502. (600 x)

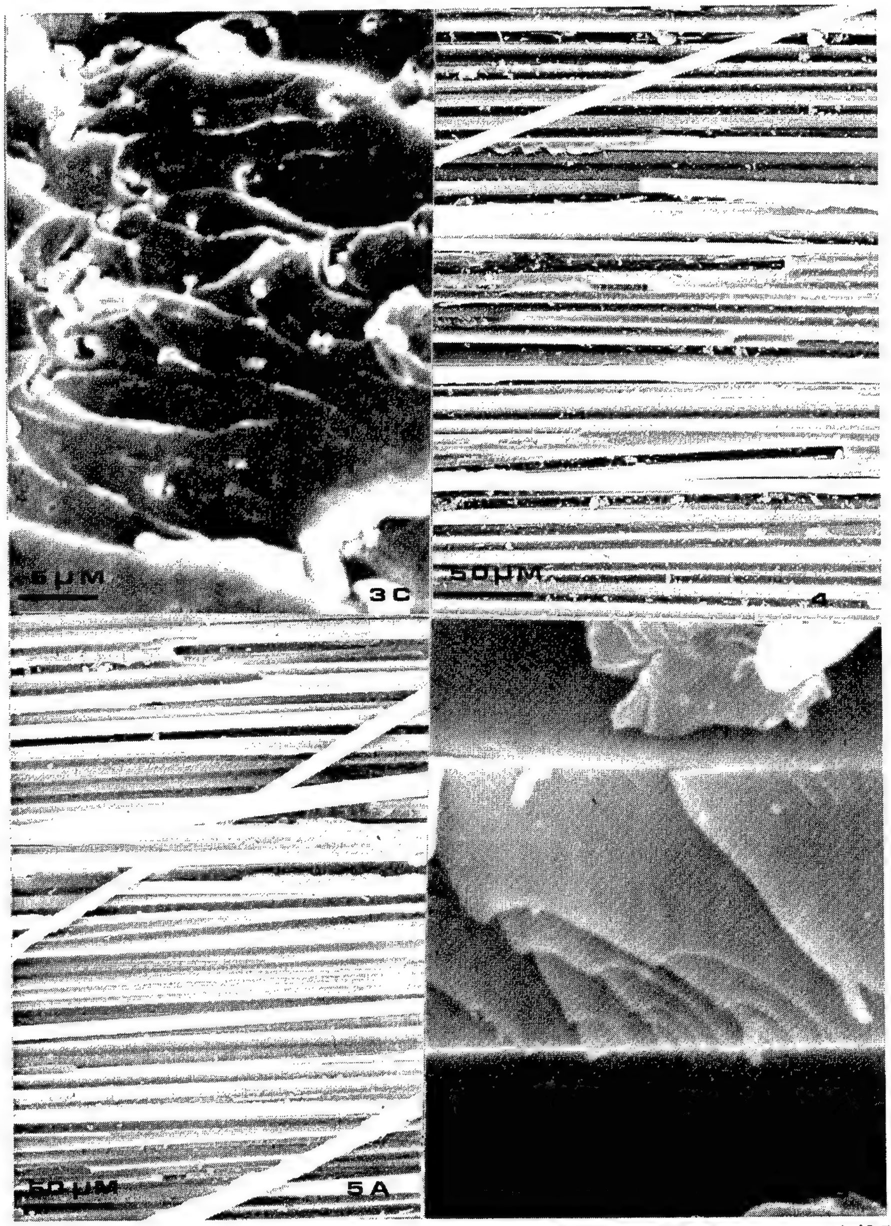




2E In-situ delamination of F155 composite (bled during curing) showing wider microcrack region than unbled material had. (1000 x)

3A In-situ transverse fracture of F155 resin system(using a miniature C.T. type specimen). (3000 x)

3B Fracture surface of specimen fractured in Figure 3A. (300 x)

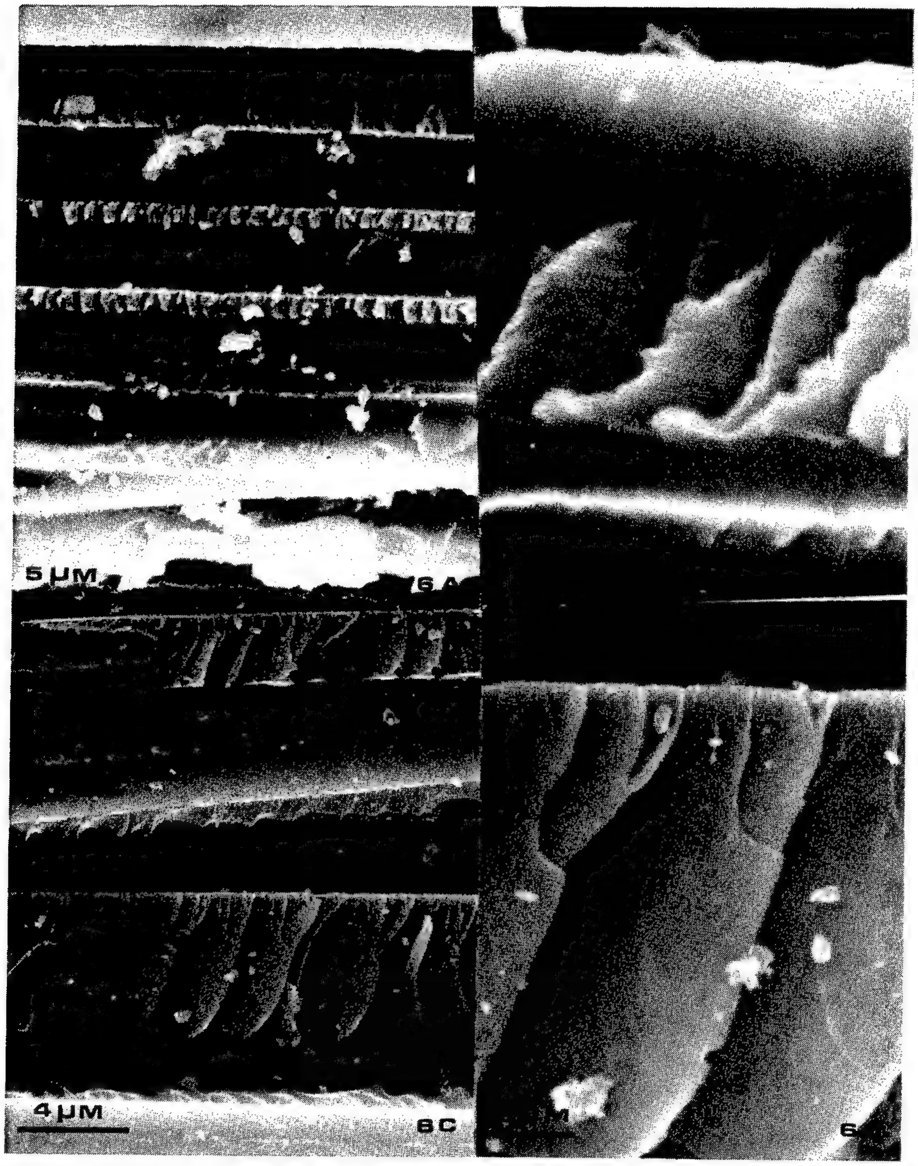


3C Fracture surface of specimen fractured in Figure 3A. (3000 x)

5A Fracture surface of AS4/3502 delaminated with 12% Mode 2 showing some resin damage. (300 x)

4 Fracture surface of AS4/3502 fractured in Mode 1. (300x)

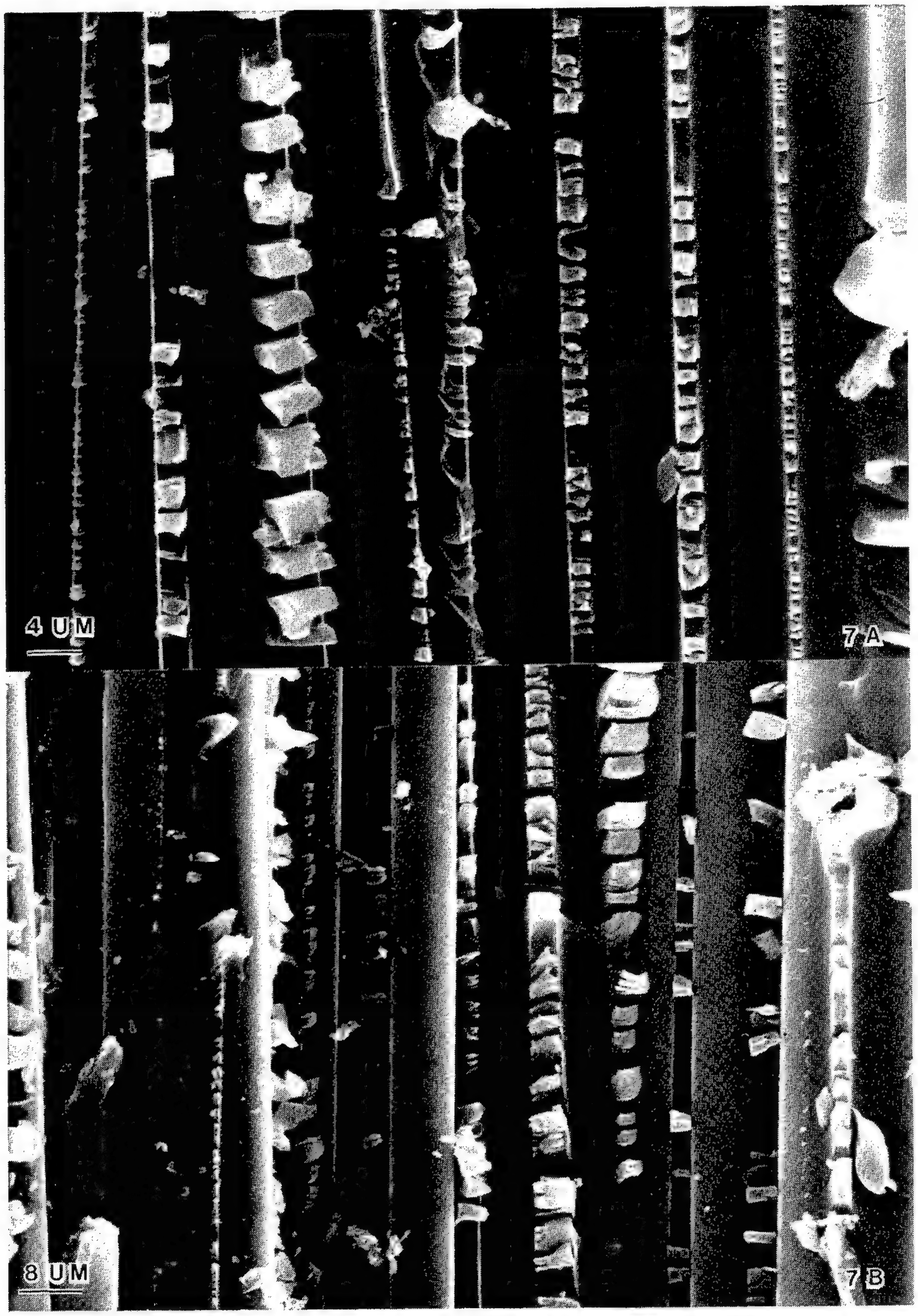
5B Fracture surface of AS4/3502, 12% Mode 2 (top surface) showing some shallow hackles pointing in crack growth direction(7000 x)



6A,6B,6C, & 6D
All four photographs show AS4/3502 fractured with 43% Mode 2 and with crack growing right to left.

Hackles with 43% Mode 2 are more numerous and sharper in angle than those in Figure 5 with 12% Mode 2.

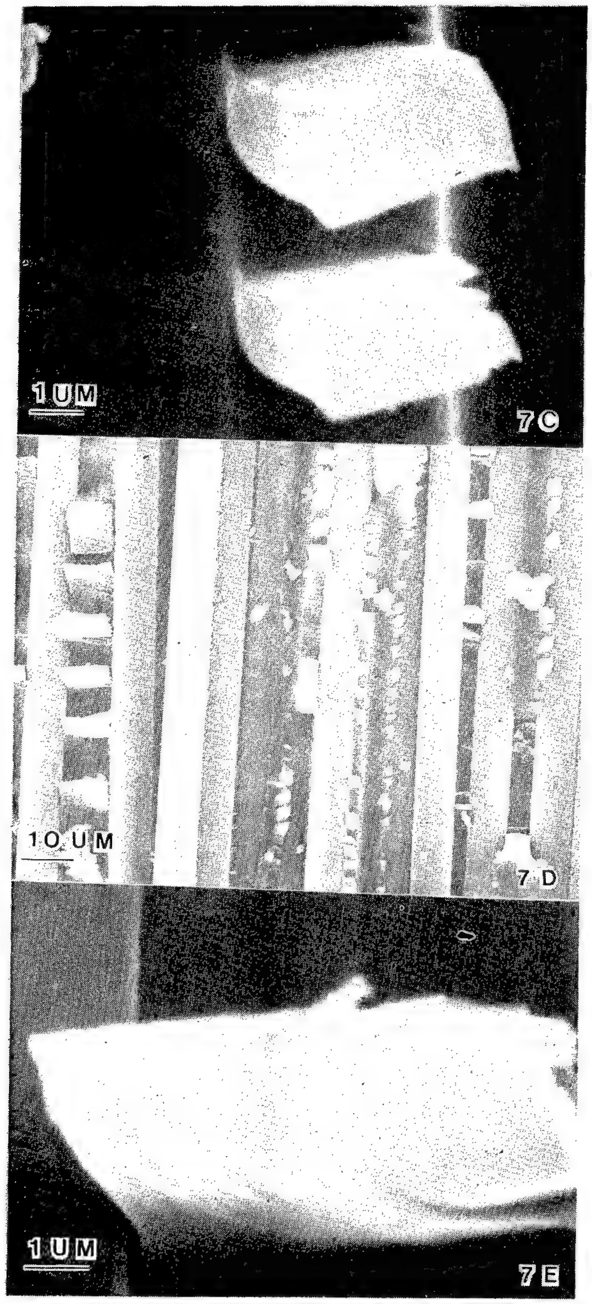
6A Top surface(2000 x)
6B Top surface (7000 x)
On top surface hackles point in direction of crack growth.
6C Bottom surface (4500 x)
6D Bottom surface (7000 x)
On bottom surface hackles point in opposite direction to crack growth.



7A & 7B
Both photographs show AS4/3502 fracture in Mode 2 and with crack growing top to bottom.
Hackles in pure Mode 2 are more numerous and sharper in angle than those in Figure 6 at 43% Mode 2.

7A Top surface(2400 x)
7B Bottom surface (1600 x)

On top surface hackles point in crack growth direction and on bottom surface they point opposite to crack growth direction.



7C, 7D, & 7E
All three photographs show AS4/3502 fractured in pure Mode 2 and with crack growing top to bottom.

Hackle angles are very sharp and in some cases approach being vertical to the surfac(7C and 7E).

7C Top surface(10,000 x)

7D Bottom surface(1000 x)

7E Bottom surface(10,000 x)

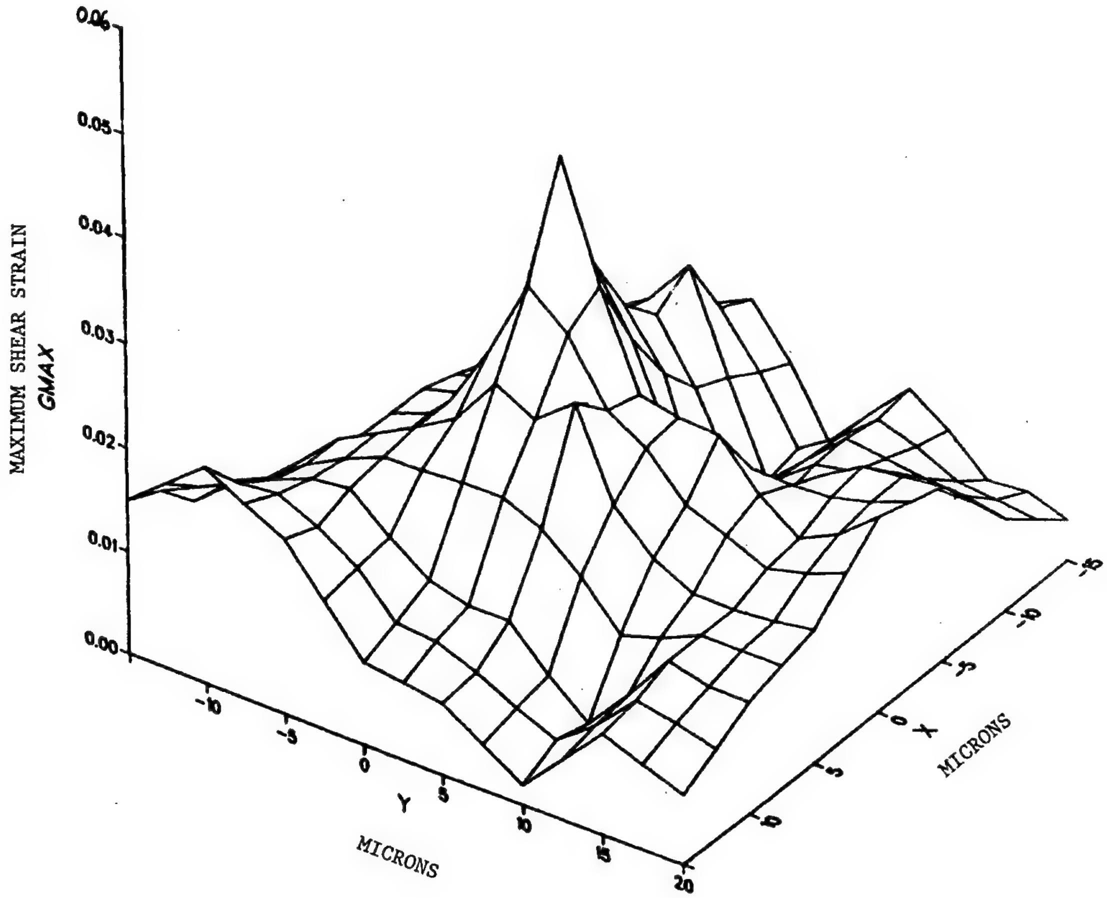


FIGURE 8. Shear strain field around crack tip at x=0, y=0, as measured using relative displacements in scanning electron microscope photographs of a single edge notched unidirectional specimen loaded perpendicular to the fiber direction.

TWO-DIMENSIONAL MODELING OF COMPRESSIVE FAILURE IN DELAMINATED LAMINATES

Herzl Chai

University of Dayton
Research Institute
Dayton, Ohio 45469

ABSTRACT

An analytical model is developed to assess the compressive strength criticality of near-surface interlaminar delamination in anisotropic laminated composites. This model is an extension of a previously developed model for the case of isotropic [1] and orthotropic [2] laminates.

The present presentation is concerned with a delaminated region which is elliptic in shape, located between a thick, isotropic plate and a thin, homogeneous and orthotropic layer whose material axes coincide with the ellipse axes. The growth conditions and growth behavior of this defect are studied by breaking the overall problem into an elastic stability problem and a fracture problem. Post-buckling solution for the elliptic section is obtained using the Rayleigh-Ritz method while a simple energy balance criterion, subject to the condition of self-similar growth, governs the fracture.

The parameters controlling the growth or arrest of the delamination damage are identified as the fracture energy, the disbond depth, and the elastic properties of the thin layer material and the thick isotropic plate.

By varying the degree of material anisotropy, a range in growth behavior was found including stable or unstable crack growth parallel to or normal to the loading axis.

REFERENCES

1. H. Chai, "The Growth of Impact Damage in Compressively Loaded Laminates," Ph.D. Thesis, California Institute of Technology (March 1982).

2. H. Chai, "On the Two-Dimensional Buckling/Delamination Model-Orthotropic Case," SM Progress Report No. 6, California Institute of Technology (August 1981).

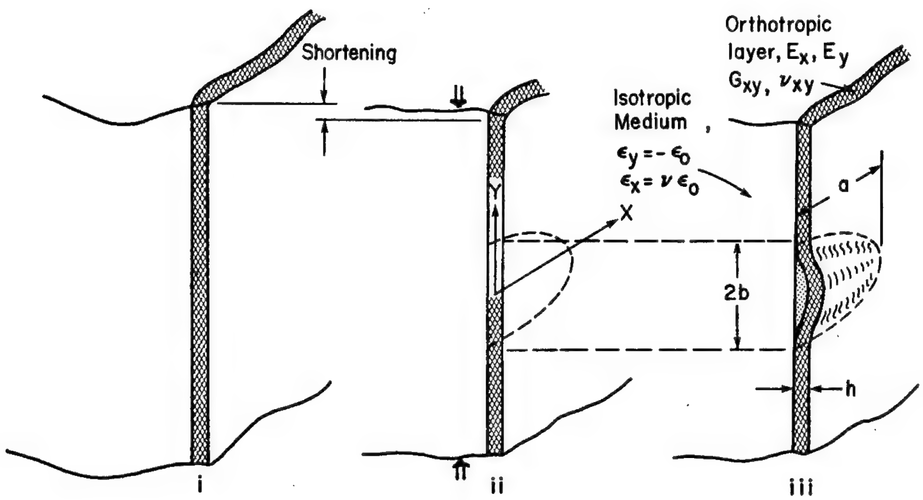


FIG. 1. 2-D DELAMINATION BUCKLING MODEL

POST-BUCKLING OF ELLIPTIC PLATE

BOUNDARY CONDITIONS ON $X^2/a^2 + Y^2/b^2 = 1$:

$$U = \nu\epsilon_0 X \ , \quad V = -\epsilon_0 Y \ , \quad W = \partial W/\partial X = \partial W/\partial Y = 0$$

ASSUMED SOLUTION:

$$\frac{bu}{h^2} = \frac{\nu\lambda\bar{x}}{m}\,[1 + (\bar{x}^2/\xi^2 + \bar{y}^2 - 1)\, c_1]$$

$$\frac{bv}{h^2} = \frac{\lambda\bar{y}}{m}\,[-1 + (\bar{x}^2/\xi^2 + \bar{y}^2 - 1)\, c_2]$$

$$\frac{w}{h} = (\bar{x}^2/\xi^2 + \bar{y}^2 - 1)^2 (c_3 + c_4\bar{x}^2 + c_5\bar{y}^2)$$

WHERE

$$\bar{x} = \frac{x}{b},\ \bar{y} = \frac{y}{b},\ \bar{u} = \frac{ub}{h^2},\ \bar{v} = \frac{vb}{h^2},\ \bar{w} = \frac{w}{h}$$

$$\xi = a/b = \text{aspect ratio} \quad , \quad m = 1 - \nu_{xy}\,\nu_{yx}$$

$\lambda = m\,\varepsilon_o\,(b/h)^2$ = load-geometry parameter

FIND $c_1 \ldots c_5$ FROM THE CONDITION OF STATIONARY POTENTIAL ENERGY.

<u>RESULTS</u>

<u>NOTATION FOR FIGURES</u>

ISOTROPIC MEDIUM - $\nu = 0.3$

THIN LAYER - T300/5208 UNIDIRECTIONAL LAMINATE:

FIBER ALIGNS VERTICALLY

FIBER ALIGNS HORIZONTALLY

THIN LAYER - ISOTROPIC, $\nu = 0.3$

—-—- BUCKLING CURVES

——— INITIATION OF DELAMINATION GROWTH

NOTE IN ALL FIGURES, UPPER AND RIGHT-HAND SIDE COORDINATES ARE PARTICULARIZATIONS OF THE NONDIMENSIONAL COORDINATES (LOWER AND LEFT HAND SIDE) TO THE SPECIFIC CASES SHOWN IN THE TABLE BELOW.

b(IN)	h(IN)	Γ(LB/IN)	E(PSI)	E_Y(PSI)	FIGURE
0.5	0.03				3
		0.5	$1x10^7$		5
	0.03	0.5	$1x10^7$		4
	0.03	0.5		$2.62x10^7$	6
	0.03	0.5		$0.15x10^7$	7

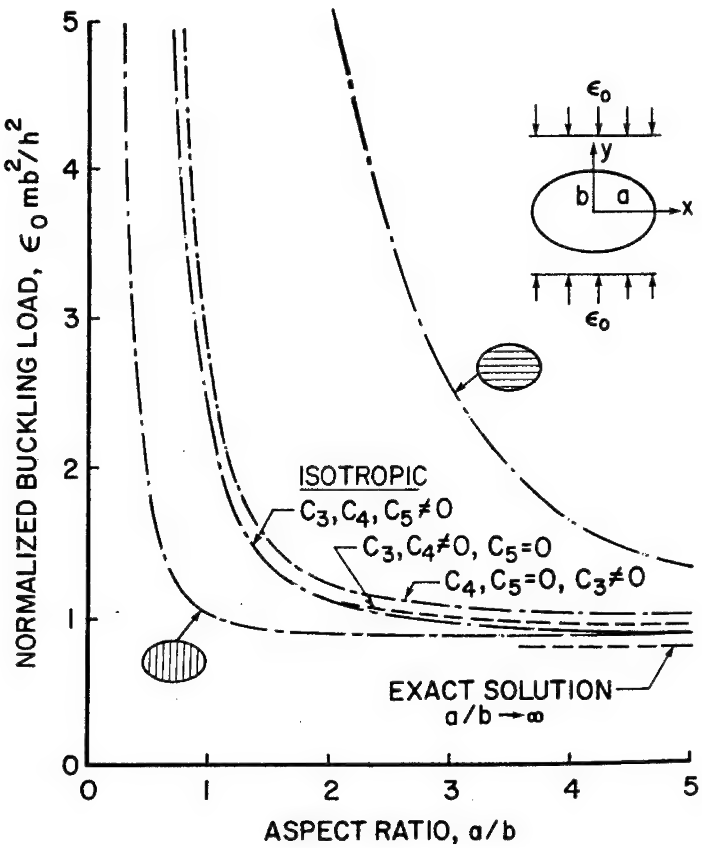


FIG. 2. BUCKLING CONDITIONS

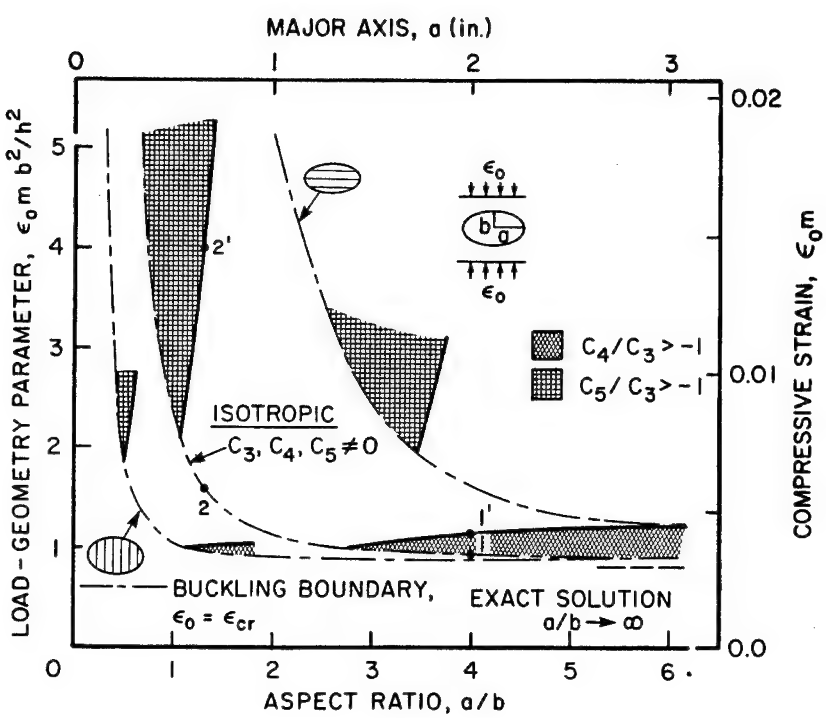
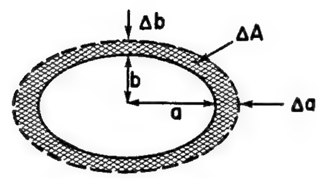


FIG. 3. POST-BUCKLING DEFLECTION (W) CONDITIONS

FRACTURE CRITERION



ASSUME SELF SIMILAR GROWTH

$$G = \frac{U(a,b) - U(a+\Delta a, b+\Delta b)}{\Delta A}, \quad \Delta A = \pi(a\Delta b + b\Delta a)$$

FOR A GIVEN UNIT AREA ΔA, FIND THE RATIO $\Delta a/\Delta b$ WHICH MAXIMIZES G. DELAMINATION GROWTH OCCURS WHENEVER

$$G \geq \Gamma = \text{FRACTURE ENERGY/UNIT AREA}$$

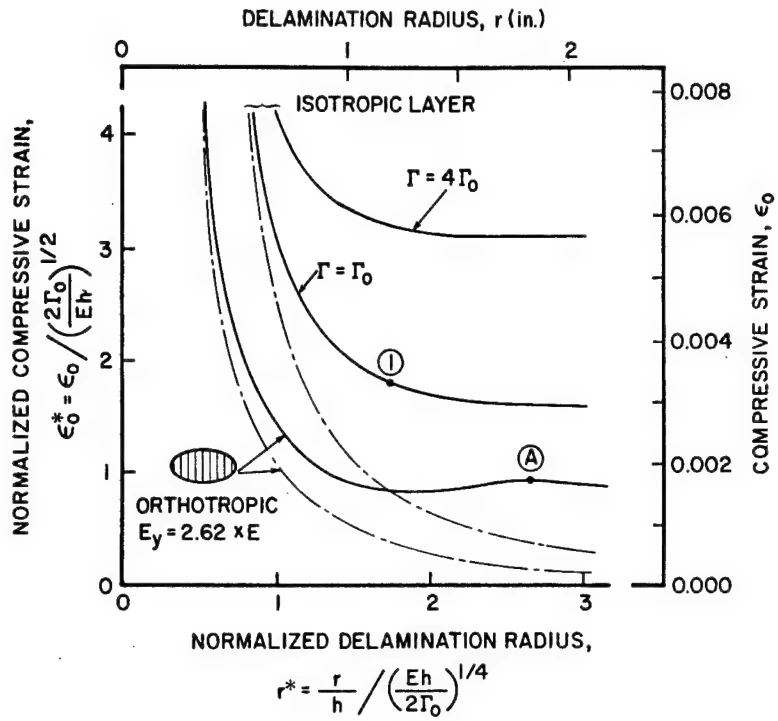


FIG. 4. BUCKLING AND GROWTH CONDITIONS FOR A CIRCULAR DELAMINATION

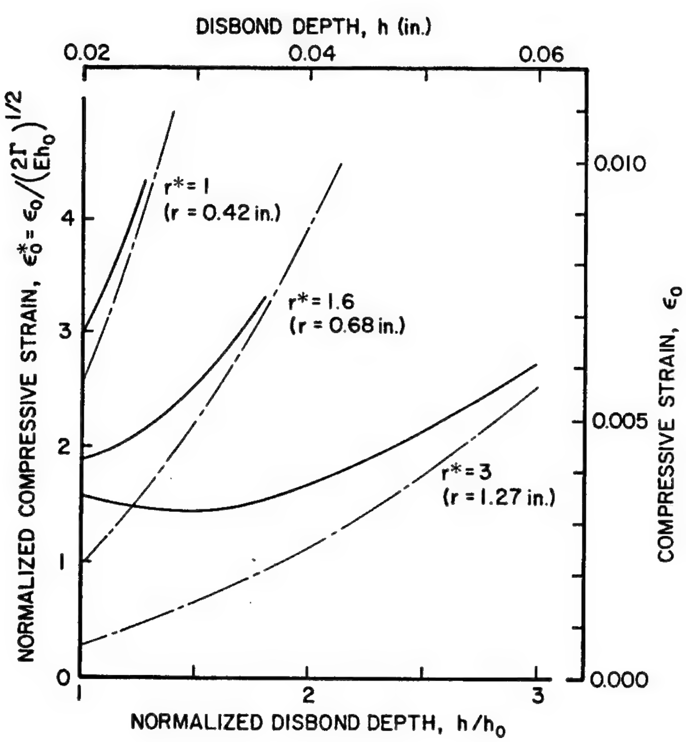


FIG. 5. EFFECT OF DISBOND DEPTH (h) ON BUCKLING AND GROWTH OF CIRCULAR DELAMINATION

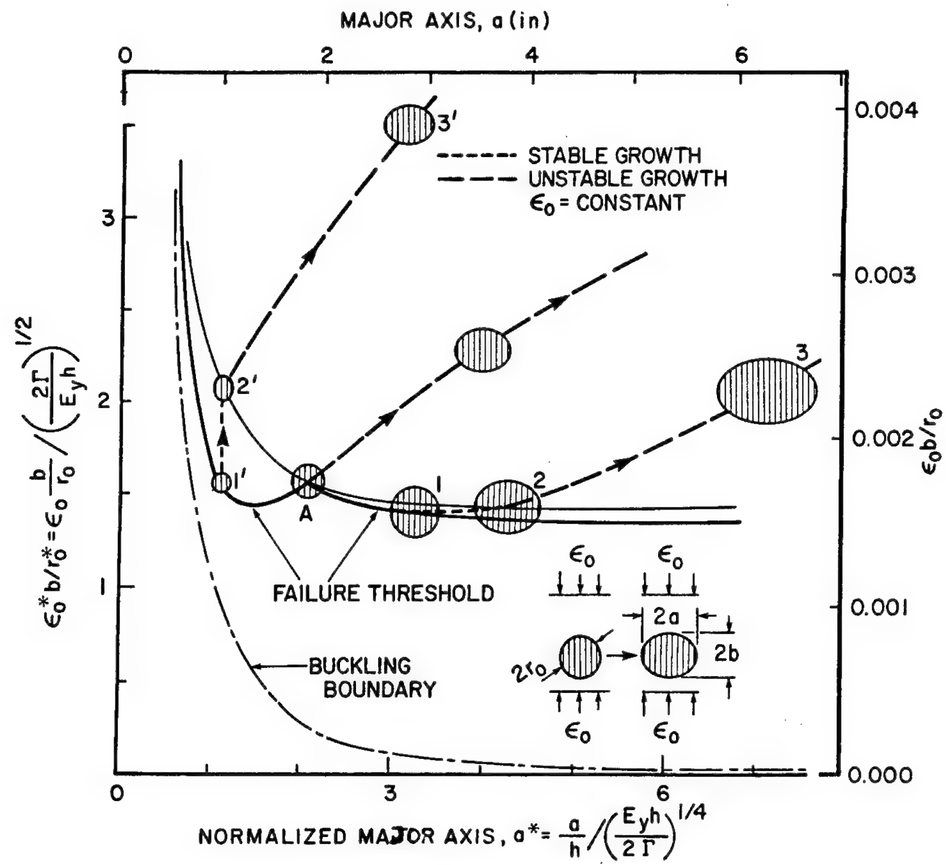


FIG. 6. BUCKLING, GROWTH INITIATION AND GROWTH BEHAVIOR FOR AN INITIALLY CIRCULAR DELAMINATION

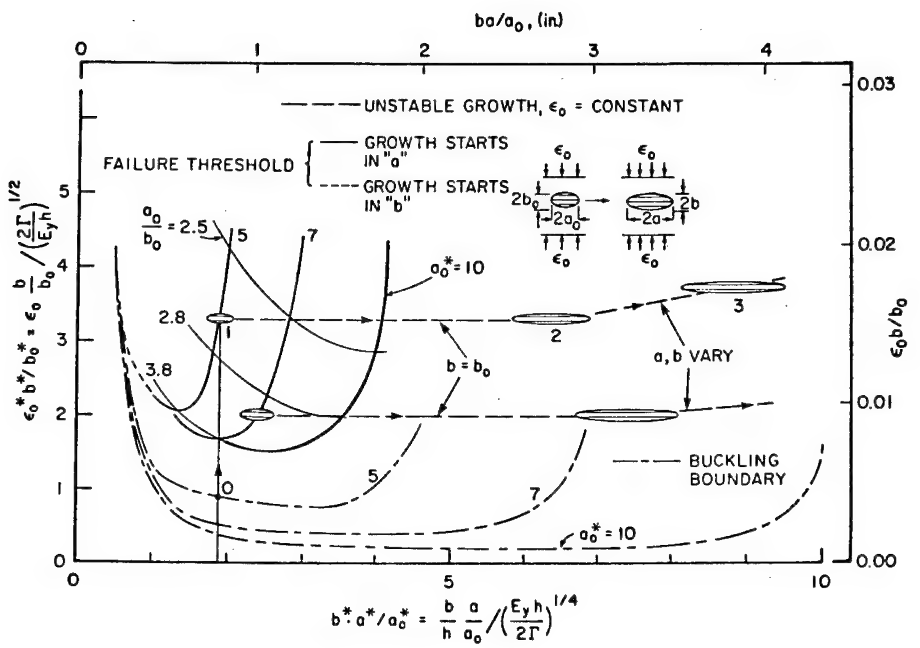


FIG. 7. AS IN FIG. 6 FOR ELLIPTIC DELAMINATION

PREDICTION OF COMPRESSION FAILURE

John D. Whitcomb

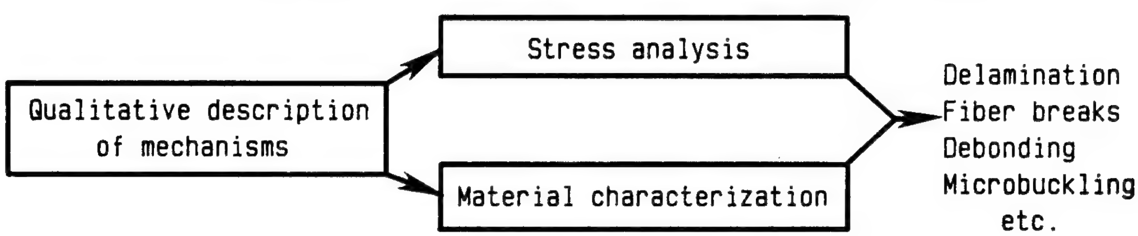
NASA Langley Research Center
Hampton, Virginia 23665

ABSTRACT

One of the research thrusts in the Fatigue and Fracture Branch at NASA Langley Research Center is the prediction of compression failure using fracture mechanics. The figure below is a flow chart of the necessary activities and the topics which are being emphasized in each activity. This talk will discuss several topics which illustrate the approach taken in addressing the needs shown. These topics are:

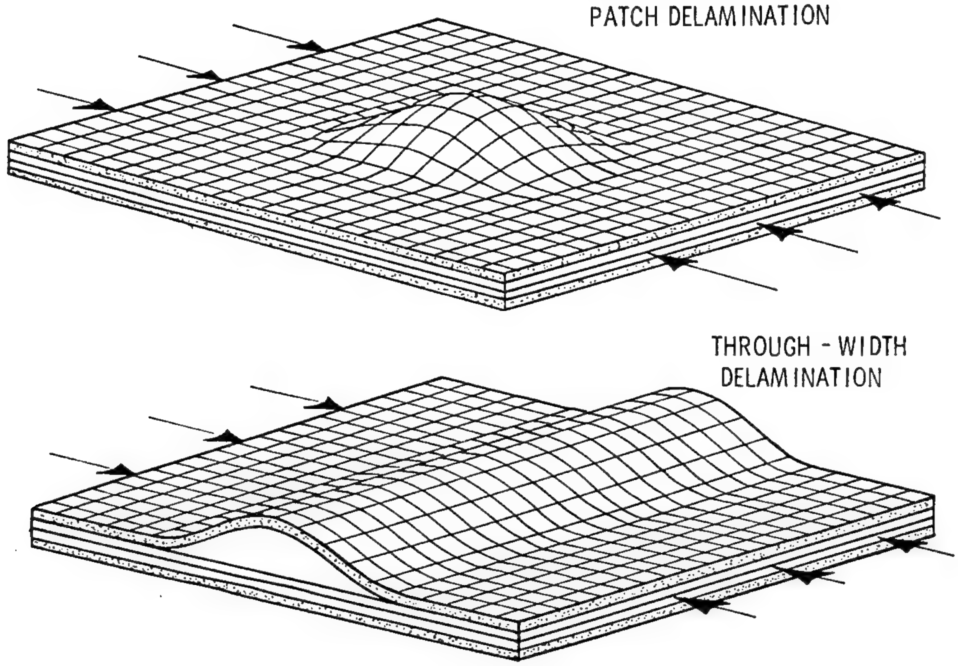
1. Instability-related delamination growth
2. Analysis of thick laminates
3. Dynamic effects on mode I interlaminar fracture toughness
4. Microbuckling

PREDICTION OF COMPRESSION FAILURE

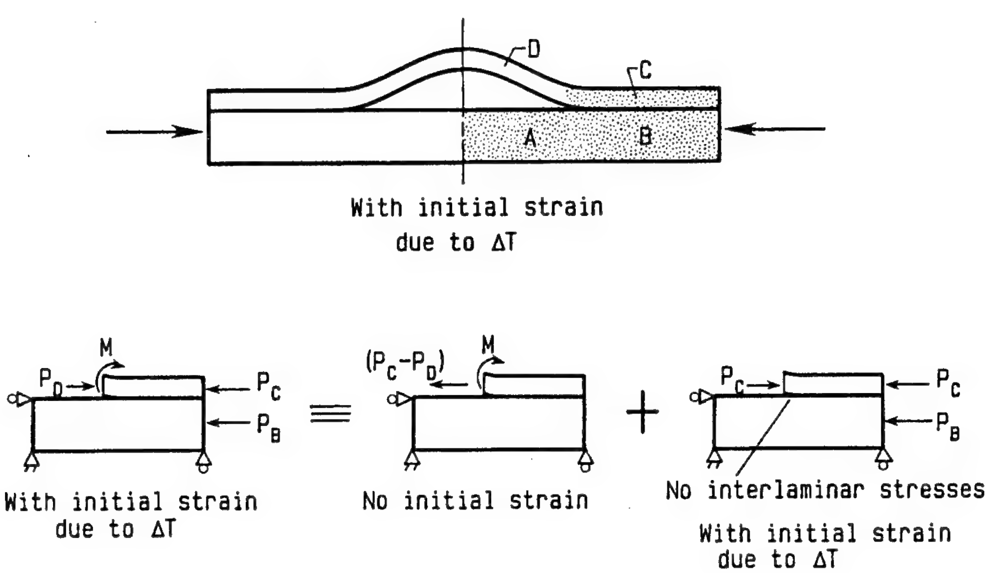


Mechanisms	Stress analysis	Material Characterization
microbuckling	thick laminates	microbuckling
lamina buckling	geometric and material nonlinearity	mixed-mode delamination
fiber failure	approximate analyses	dynamic effects
delamination		
ply cracking		
synergism		

LOCAL BUCKLING OF DELAMINATED PLIES

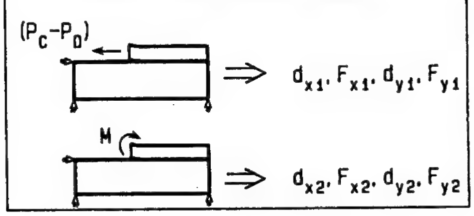


APPROXIMATE ANALYSIS

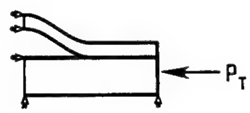


CALCULATION OF STRAIN-ENERGY RELEASE RATES

UNIT LOAD SOLUTIONS



STRENGTH OF MATERIALS ANALYSIS



$$P_C-P_D = \frac{\pi^2}{16a^2}\,\frac{S_A S_D}{S_A+S_D}\,(\delta^2+2\delta\delta_0)$$

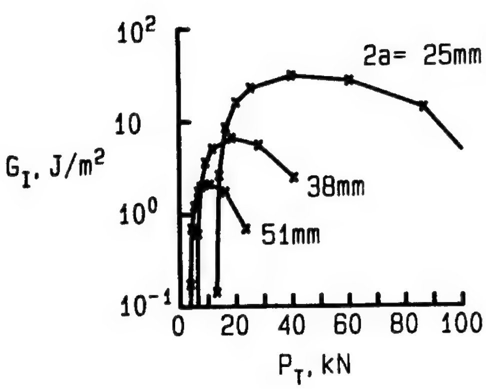
$$M = .5P_D\,(\delta+\delta_0)$$

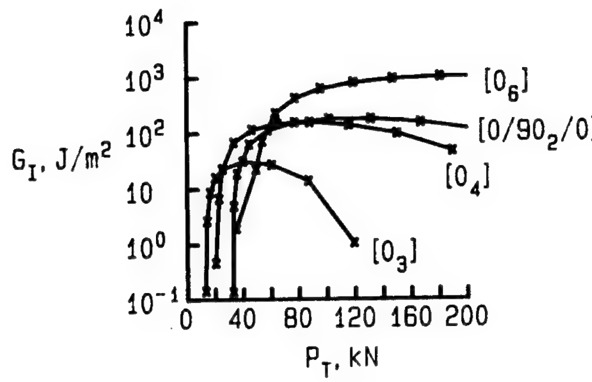
STRAIN ENERGY RELEASE RATES

$$G_I = \frac{1}{2\Delta ab}\,\frac{d_{y1}}{F_{y1}}\left[(P_C-P_D)F_{y1} + MF_{y2}\right]^2$$

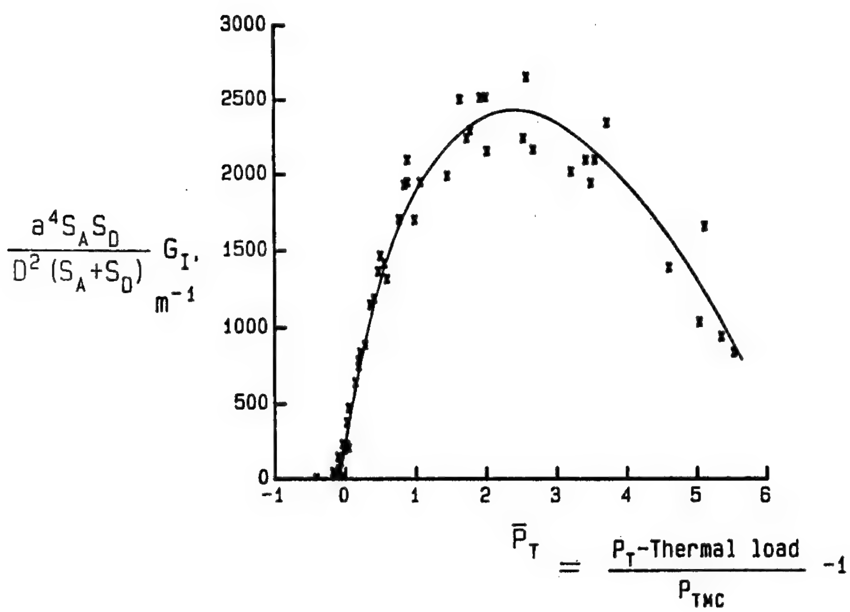
$$G_{II} = \frac{1}{2\Delta ab}\,\frac{d_{x1}}{F_{x1}}\left[(P_C-P_D)F_{x1} + MF_{x2}\right]^2$$

EFFECT OF DELAMINATION LENGTH AND LAMINATE TYPE ON G_I

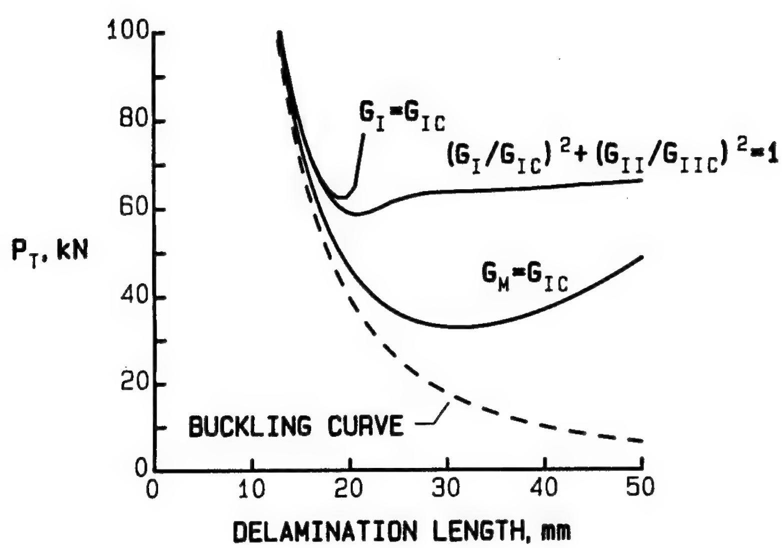




NORMALIZED G_I vs. NORMALIZED LOAD



PREDICTED LOADS FOR DELAMINATION GROWTH
INITIAL IMPERFECTION= 0



DELAMINATION UNDER COMPRESSION LOADS

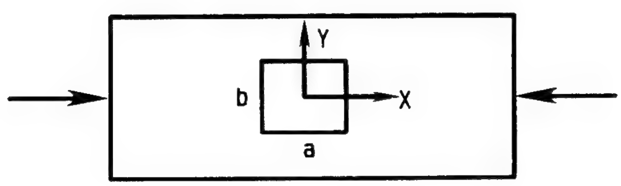
Dr. Byron Pipes
Jack Gillespie, Jr.
(U. of Delaware)

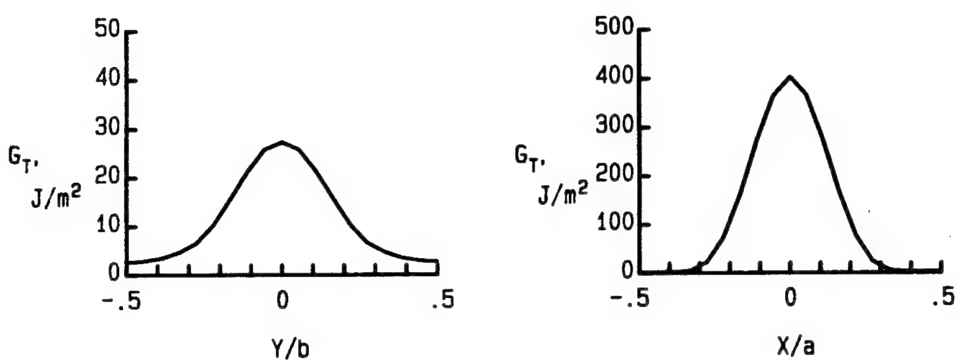
* Through-width delamination specimen

* Parameters: delamination length
  delamination location
  initial imperfection
  interlaminar fracture toughness

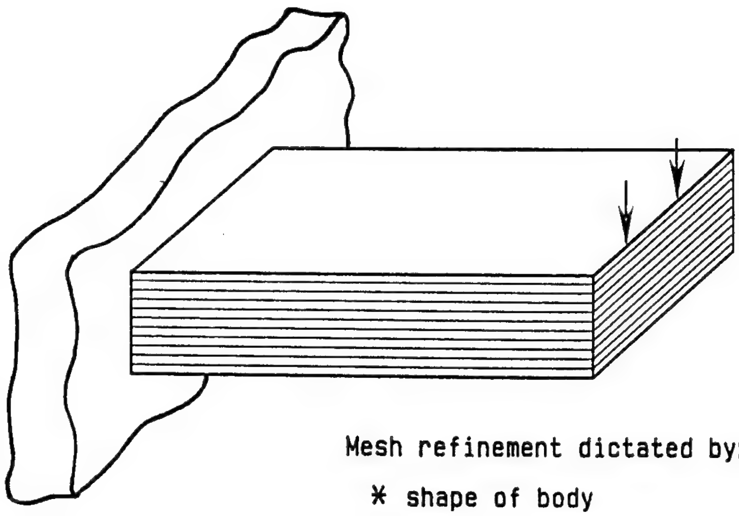
* Experiments and analysis

DISTRIBUTION OF G_T FOR A POSTBUCKLED SUBLAMINATE





ANALYSIS OF THICK LAMINATES



Mesh refinement dictated by:

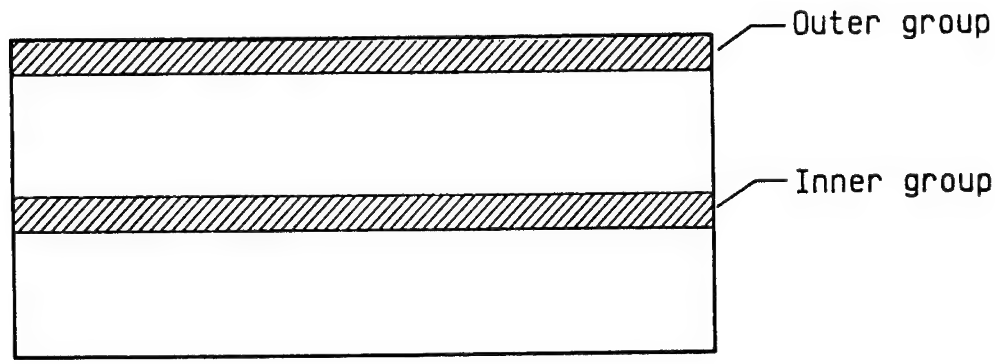
* shape of body
* strain gradients
* material property discontinuities

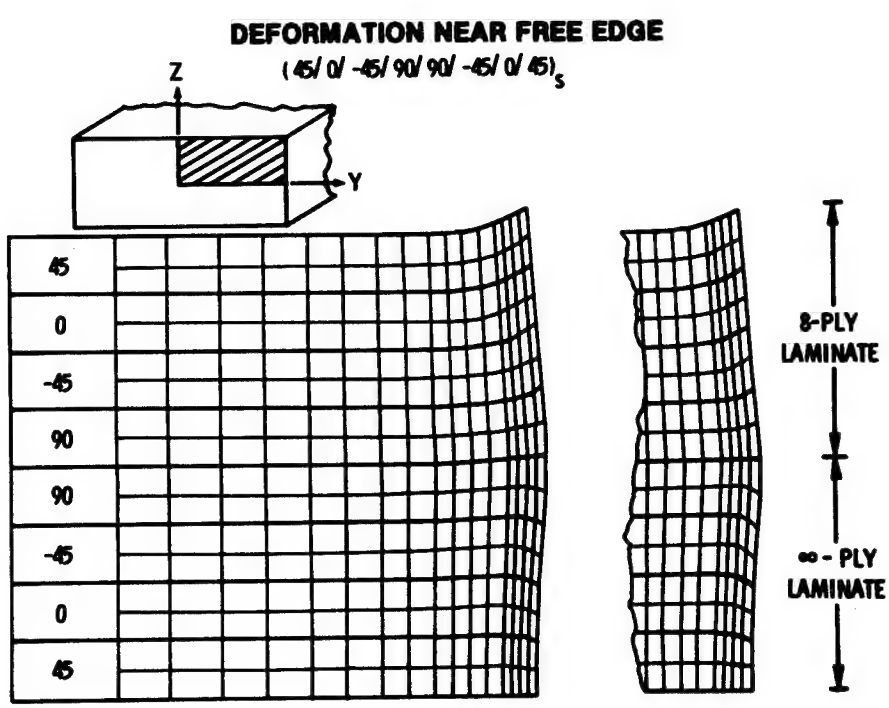
STRATEGIES FOR ANALYSIS OF THICK LAMINATES

* Analysis of representative ply groups

* Development of new type of finite element

ANALYSIS OF REPRESENTATIVE PLY GROUPS





DELAMINATION IN THICK LAMINATES

* G_T and G_I/G_T larger near surface than in interior

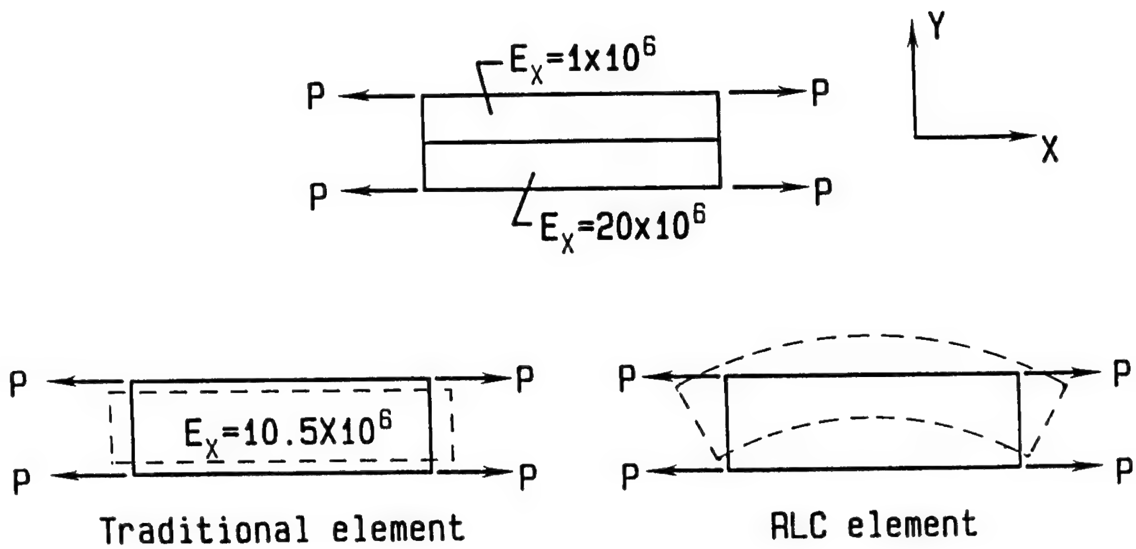
* G_T smaller for thick laminates

* Less notch blunting

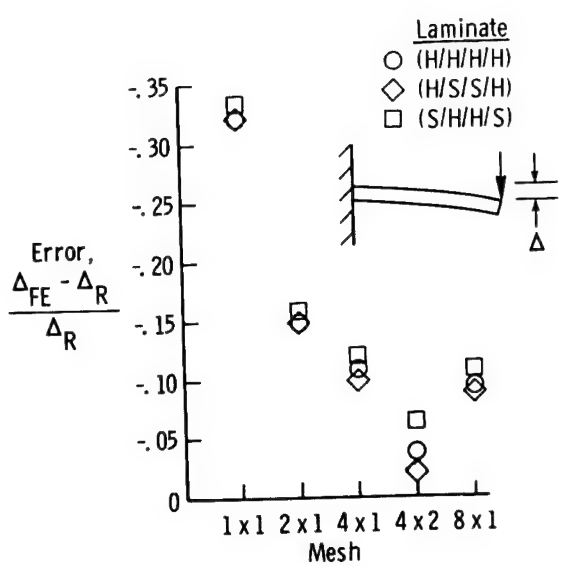
RECTANGULAR LAMINATED COMPOSITE ELEMENT

* Plane stress/plane strain
* Includes stacking sequence effects
    * Extensional and shear stiffness
    * Flexural stiffness
    * Extension-flexure coupling
* Technique valid for 3-D element

COMPARISON OF TRADITIONAL AND RLC ELEMENT



CANTILEVERED BEAM WITH TIP LOAD



EFFECTS OF STRAIN RATE ON DELAMINATION FRACTURE TOUGHNESS

Dr. I.M. Daniel
(I.I.T)

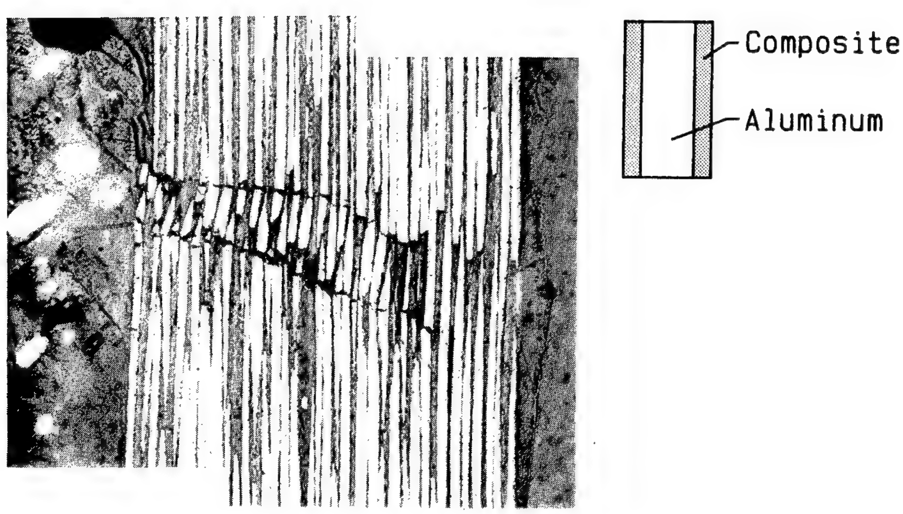
* Double cantilever beam

* Two materials: T300/3501-6 and T300/F185

* Rate of growth : quasi-static to 50mm/sec.
  (higher rates planned)

* Rate effect: T300/3501-6 -> small increase in G_{IC}
  T300/F185 -> small decrease in G_{IC}

MICROBUCKLING



CUMULATIVE DAMAGE MODEL FOR ADVANCED COMPOSITE MATERIALS

P. C. Chou
Dyna East Corporation
227 Hemlock Road
Wynnewood, PA 19096

A. S. D. Wang
Drexel University
32nd and Chestnut Streets
Philadelphia, PA 19104

ABSTRACT

The phenomena of intraply and interply matrix cracking in structural laminates have been treated as stable crack propagation/arrestment problems. The macroscopic crack growth behaviors under quasi-static loading conditions can be described by a fracture mechanics approach based on the strain energy release rate concept [1, 2]. The fundamental assumption in that approach is to assume that intraply and interply matrix failures are caused by inherent material flaws and/or man-made defects. These then grow and/or stop under the applied loads.

It is thus required that material flaws be adequately represented and characterized as basic material properties; and the individual growth mechanisms as well as their mutual interactions be analytically described by an adequate modeling technique.

As it will be shown in this presentation, intraply matrix cracking generally involves crack-growth and crack-arrest in a given material layer, resulting in a random multiple-cracks formation process during the course of loading. As for interply cracking, it occurs only near locations where high interlaminar stresses are present; growth of the local flaws under sustained loading then leads to localized interply delamination.

To model the growth behavior of these two modes of matrix cracking, the concept of effective flaw distribution is applied to represent the inherent material flaws. Stress analysis based on ply-elasticity is employed to compute the individual flaw-field stresses and the associated energy release rates. Fracture mechanics criteria are used to describe the growth and growth stability of the individual flaws. And, a method of stochastic processes is developed for the random multiple-cracks formation process during the course of loading.

Using the same energy release rate as crack growth driving force for individual crack growth under cyclic loads, a crack-growth rate law is employed. For example, let a denote the area of an interply delamination, and Γ denote the plane contour of the delamination. Then by a ply-elasticity-based finite element routine, the available strain energy release rate $G(\Gamma, \sigma)$ can be computed for the delamination crack under the cyclic load amplitude σ. The proposed delamination growth rate equation takes the form

$$\frac{da}{dN} = \alpha[G(\Gamma, \sigma)/G_c]^p \qquad (1)$$

where G_c is the material fracture toughness against the considered delamination; N is the number of fatigue cycles; and α, p are fatigue parameters dependent on the material system only.

Equation (1) can be integrated to obtain $a = a(\sigma, N)$; or, treating σ as a parameter, a family of $a = a(N)$ curves for a range of constant-σ values.

Alternatively, Equation (1) also yields a family of curves of $\sigma = \sigma(N)$ for a range of constant-a values. These constant-a (or constant-damage) curves then form the basis for cumulative delamination growth under variable amplitude fatigue loads.

The energy approach has been applied to laminates that are concurrently tested in an experimental study. The laminates tested are in the form of tensile coupons made from AS-3501-06 graphite-epoxy unidirectional tape. Three sets of laminates are experimented: (a) $[0_2/90_2]_s$ and $[0_2/90_3]_s$ for intraply transverse cracks; (b) $[\pm 25/90_n]_s$, n = 1, 2, 3 for free-edge induced interply delamination; and (c) $[\pm 45/0/90]_s$, $[\pm 45_2/0_2/90_2]_s$ and $[\pm 45/0/90]_{2s}$ for both types of matrix cracks.

This presentation will highlight the essential features of the energy approach and present results from several test cases in which the analysis and experiment are compared.

REFERENCES

[1] P. C. Chou, A. S. D. Wang and H. Miller, "Cumulative Damage Model for Advanced Composite Materials," Phase-I Final Report, AFWAL-TR-82-4083, September, 1982.

[2] A. S. D. Wang, P. C. Chou and C. S. Lei, "Cumulative Damage Model for Advanced Composite Materials," Phase-II Final Report, AFWAL-TR-82-4004, March, 1984.

CUMULATIVE DAMAGE MODEL FOR ADVANCED COMPOSITE MATERIALS

P C Chou	A S D Wang
Dyna East Corp.	Drexel Univ.
227 Hemlock Rd.	32 & Chestnut Sts.
Wynnewood Pa.	Phila. Pa.

(Work Sponsored by AFWAL)

OBJECTIVE

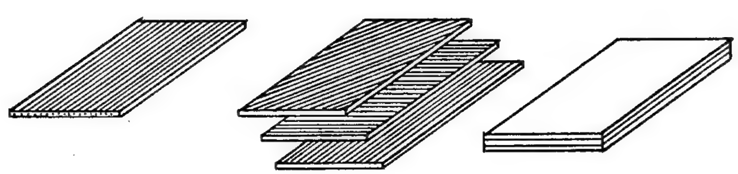
To develop an analytical model which can describe the internal matrix damage accumulation process in graphite-epoxy laminates subjected to arbitrary service loading histories

By knowing the state of damage in the laminate the residual strength, stiffness & fatigue life properties of the laminate can be estimated.

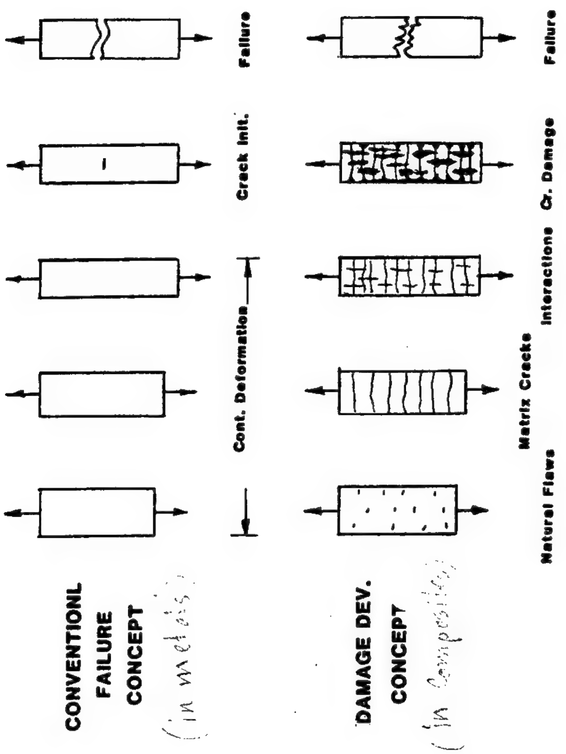
APPROACH

- o Distinguish INTRAPLY and INTERPLY matrix cracks as two fundamental cracking modes
- o Identify & characterize material flaws as sources of matrx cracks: EFFECTIVE FLAW DISTRIBUTION
- o Apply fracture mechanics criteria(static) & a growth rate law(fatigue) for sin le flaw growth
- o Apply a stochastic procedure for multiple cracks formation processes
- o Apply the CONSTANT DAMAGE concept for damage accumulation under arbitrary loads
- o Laboratory experimemt to test the validity of the analytical approach.

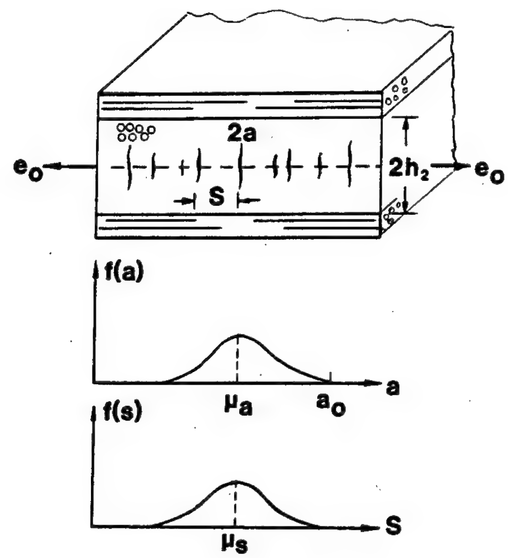
Intraply & Interply Failure Modes



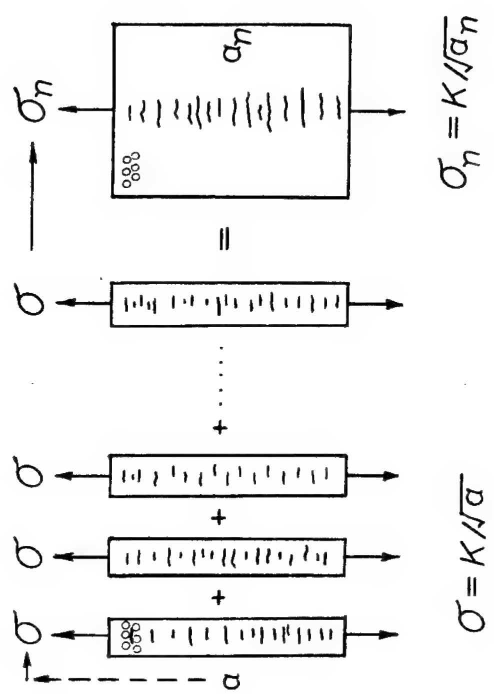
U D Ply   Lamination   Laminate



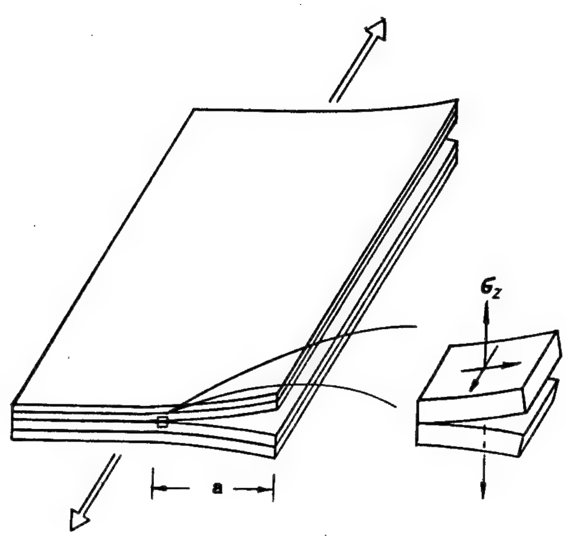
Effctive Flaws Distribution

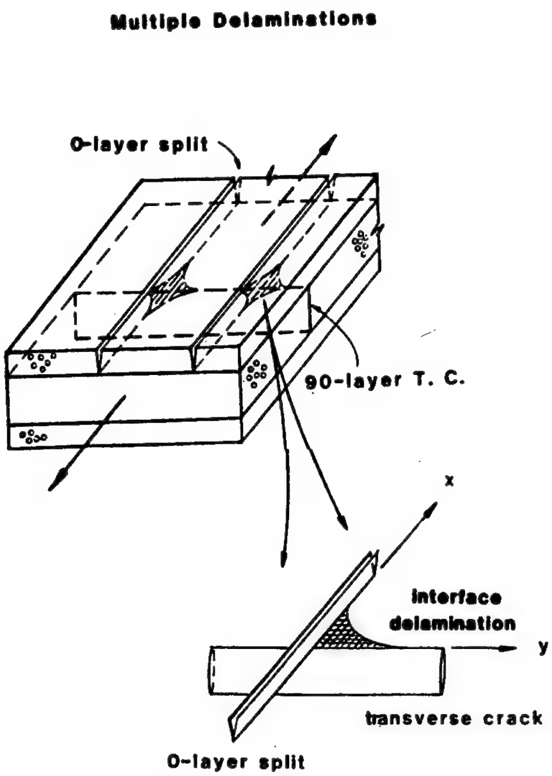


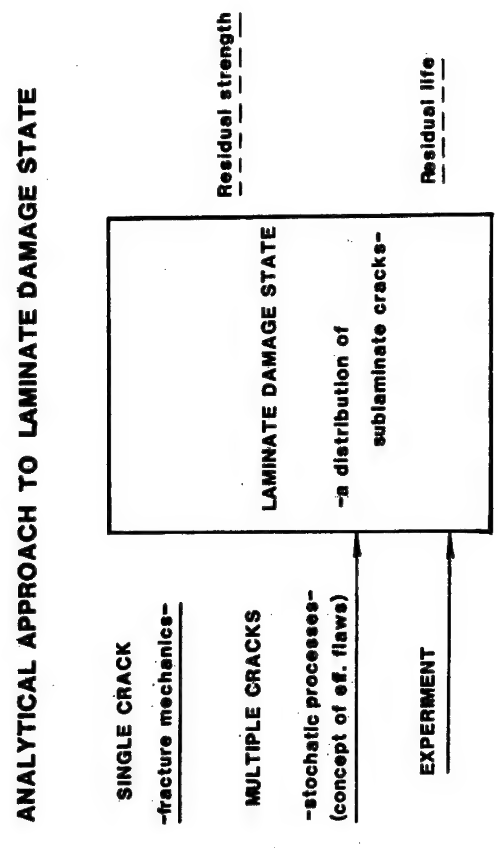
STATISTICS OF EFFECTIVE FLAWS IN 90 PLIES

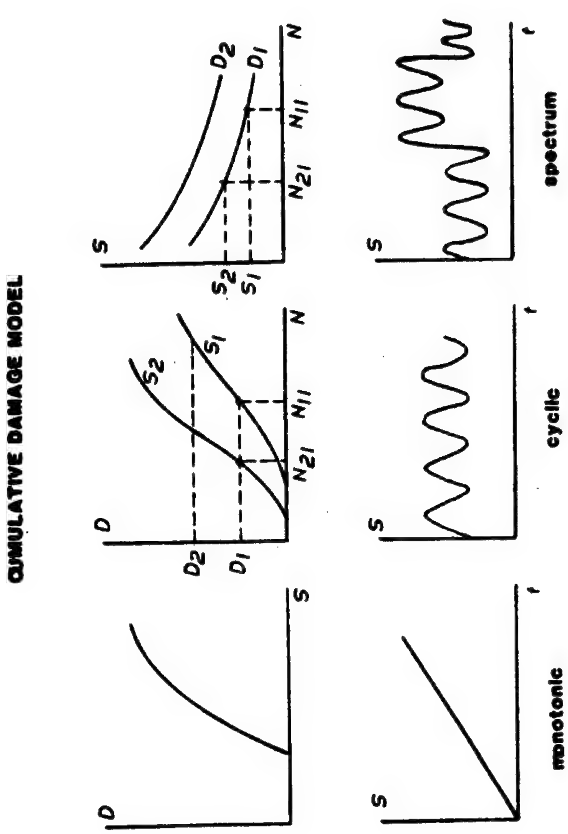


Interply Delamination









THE ENERGY MODEL

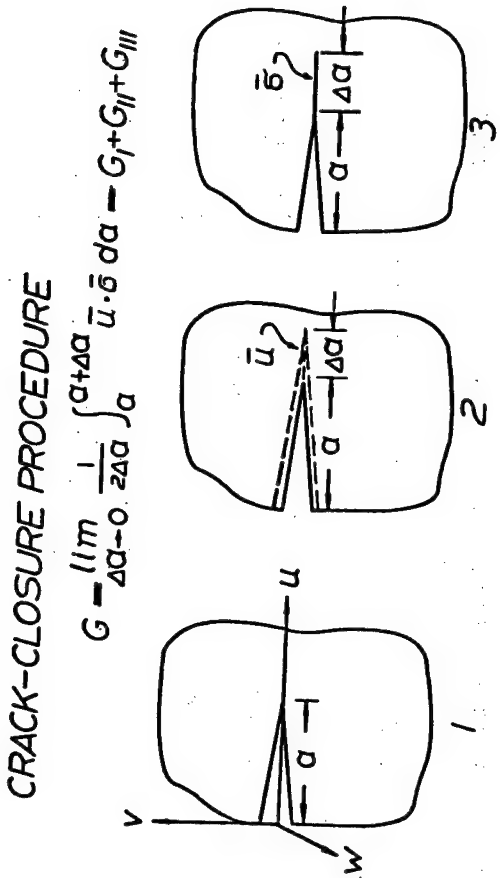
- *Judicial use of the Griffith Criterion:*

$$G = G_c$$

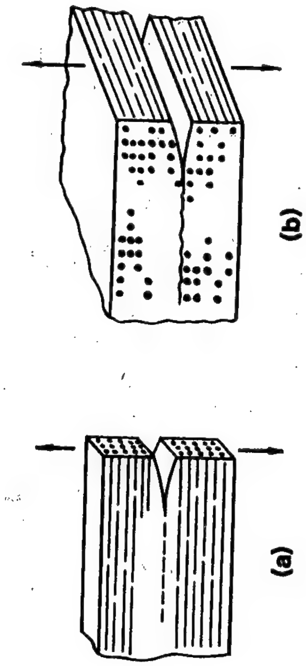
$$G(a, \underline{\bar{\sigma}, \Delta T, C_{ij}, lam.\ geo.})$$

- *Three areas of difficulty :*

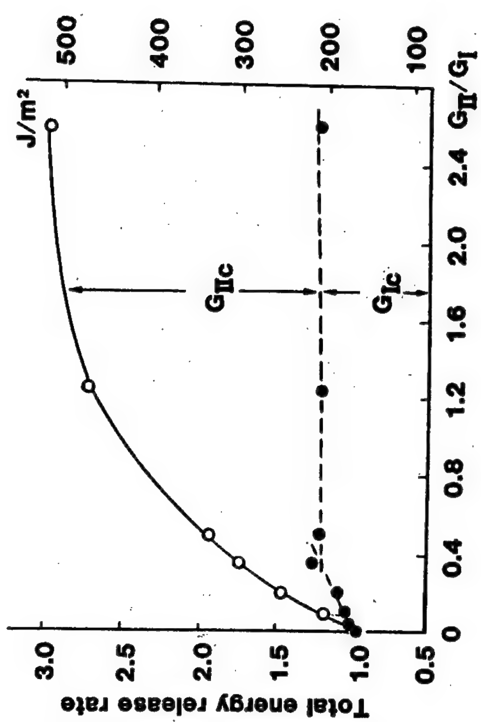
1. $a = ?$ *Random varible*
2. $G_c = ?$ *Locally dependent*
3. $G(a, \bar{\sigma}, \cdots) = ?$ *Complex geometries*



G_c Directional Dependent



$G_{c,total}$ Mixed-mode Dependent



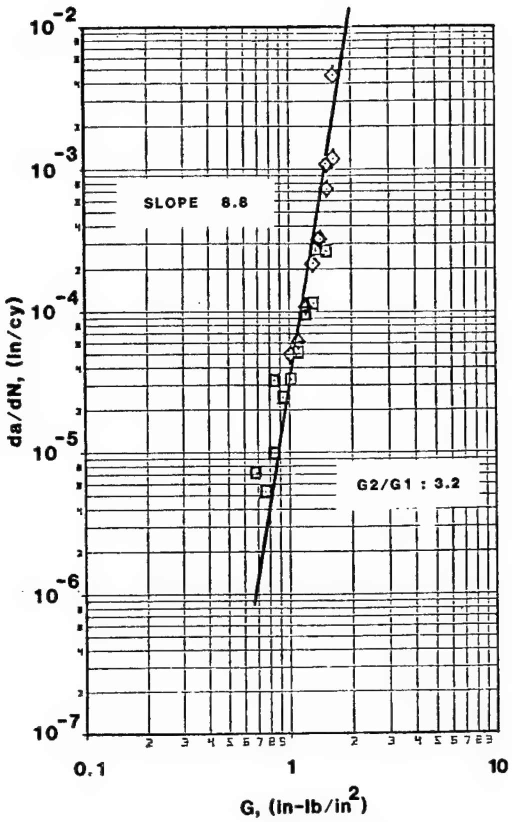
- *FATIGUE GROWTH RATE*

$$\frac{da}{dN} = \bar{\alpha}[G/G_c]^{\bar{p}}$$

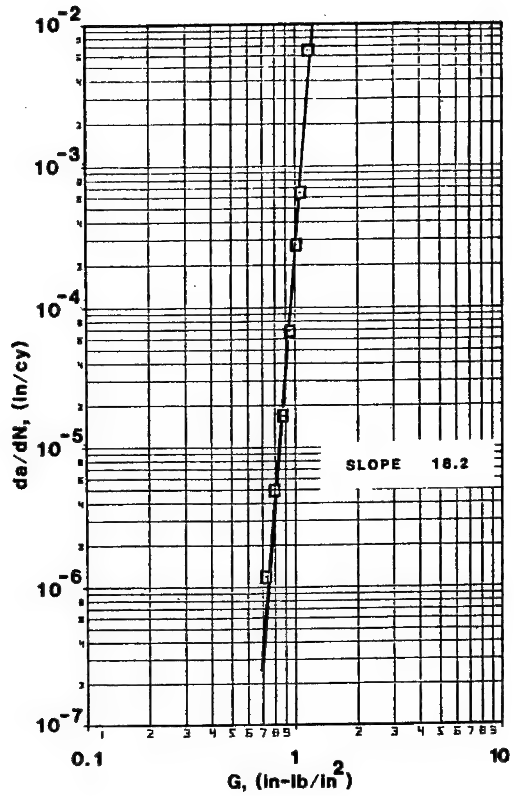
$$G = G(\bar{e}_x, \Delta T, a, geo.)$$

$\bar{\alpha}, \bar{p}$ matl. conts. ?

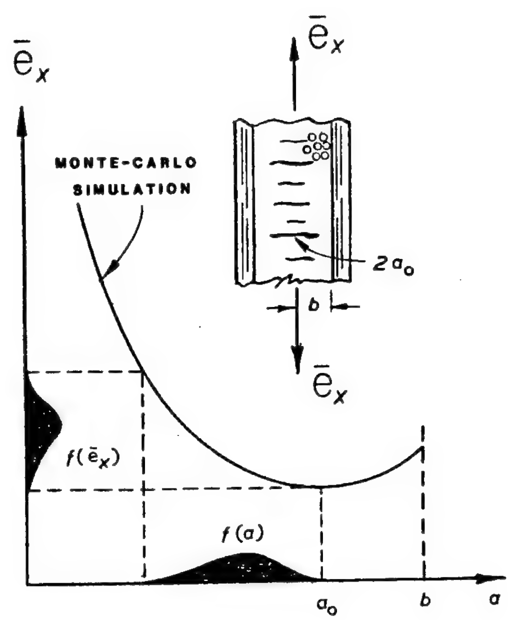
$\bar{e}_x = F(t)$; $N = f \cdot t$



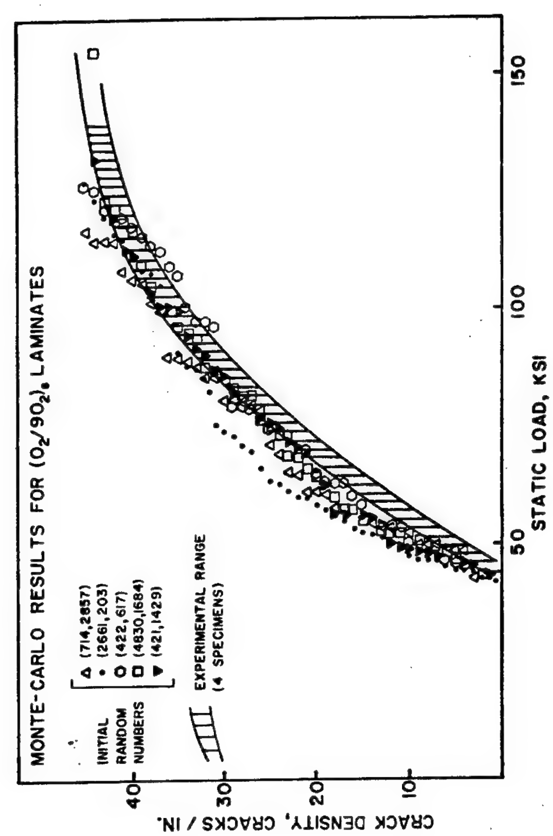
MIXED MODE GROWTH

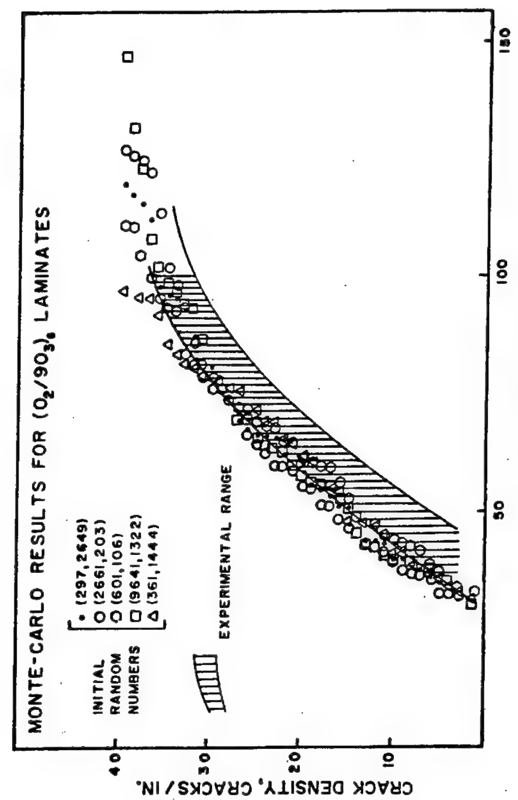


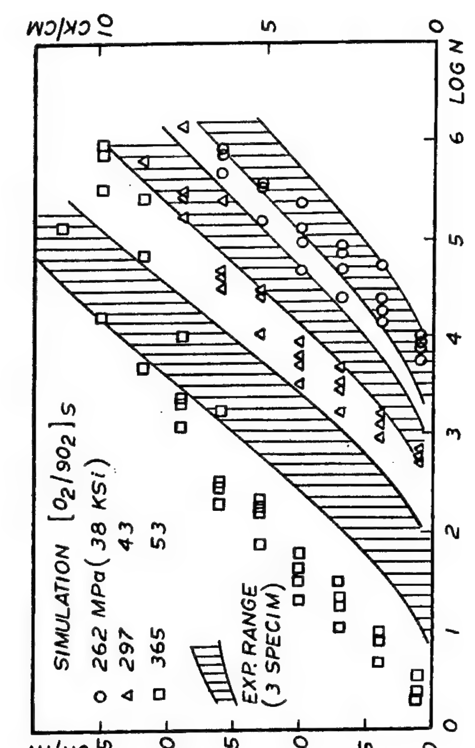
MODE I GROWTH

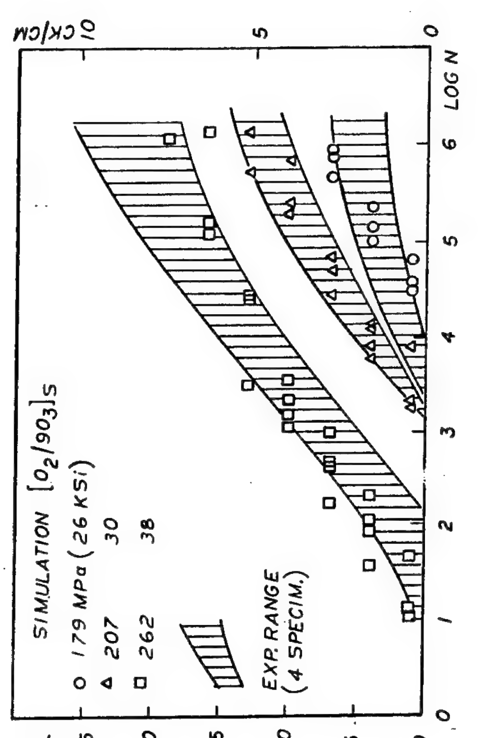


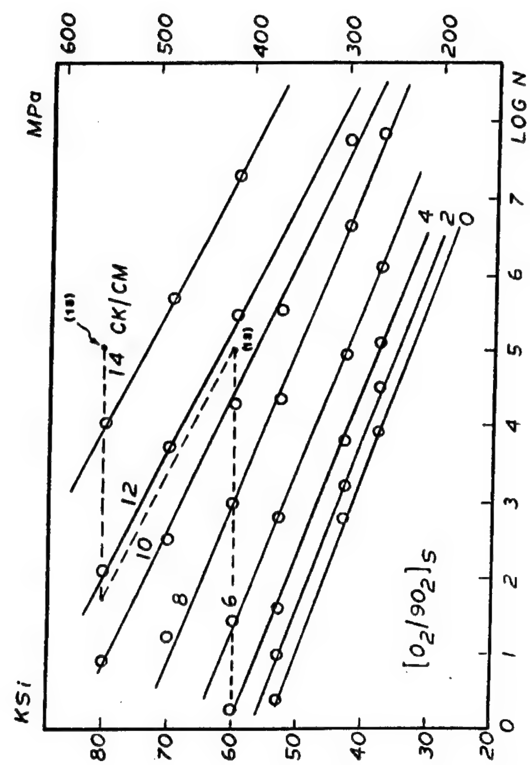
MODEL FOR MULTIPLE CRACKS



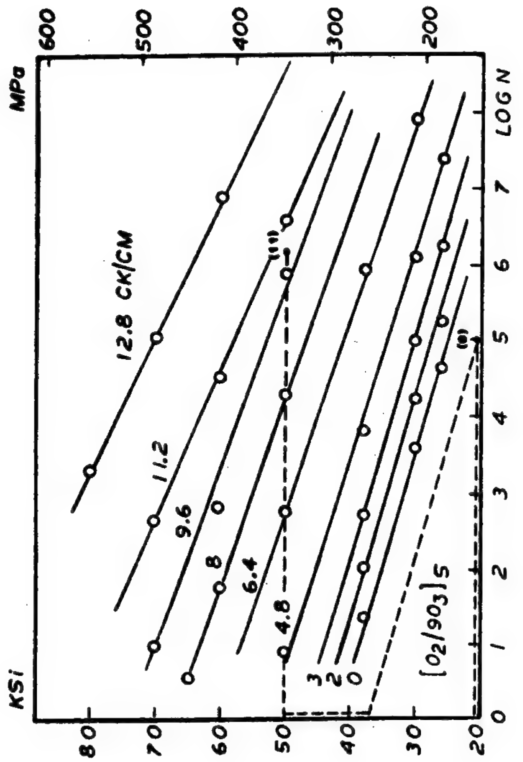

MONTE-CARLO RESULTS FOR $(0_2/90_3)_s$ LAMINATES
INITIAL RANDOM NUMBERS
• (297,2649)
○ (2661,203)
○ (601,106)
□ (9641,1322)
△ (361,1444)
EXPERIMENTAL RANGE
CRACK DENSITY, CRACKS/IN.
STATIC LOAD, KSI

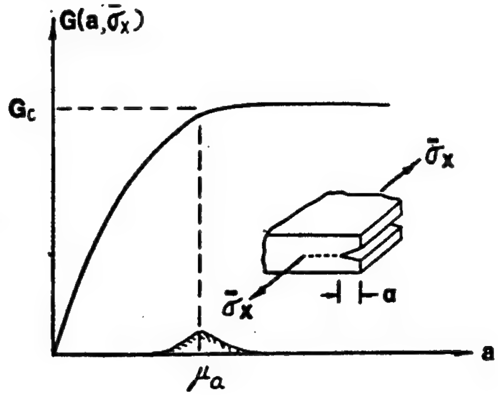

SIMULATION $[0_2/90_2]_S$
○ 262 MPa (38 KSi)
△ 297 43
□ 365 53
EXP. RANGE (3 SPECIM)
CK/IN
CK/CM
LOG N


SIMULATION $[0_2/90_3]_S$
○ 179 MPa (26 KSi)
△ 207 30
□ 262 38
EXP. RANGE (4 SPECIM.)
CK/IN
CK/CM
LOG N

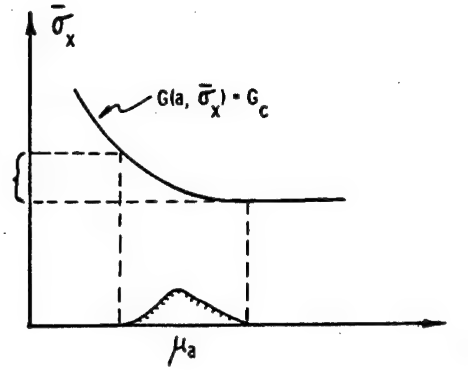

KSi
MPa
CK/CM
$[0_2/90_2]_S$
LOG N

PREDICTIVE MODEL FOR FREE EDGE DELAMINATION

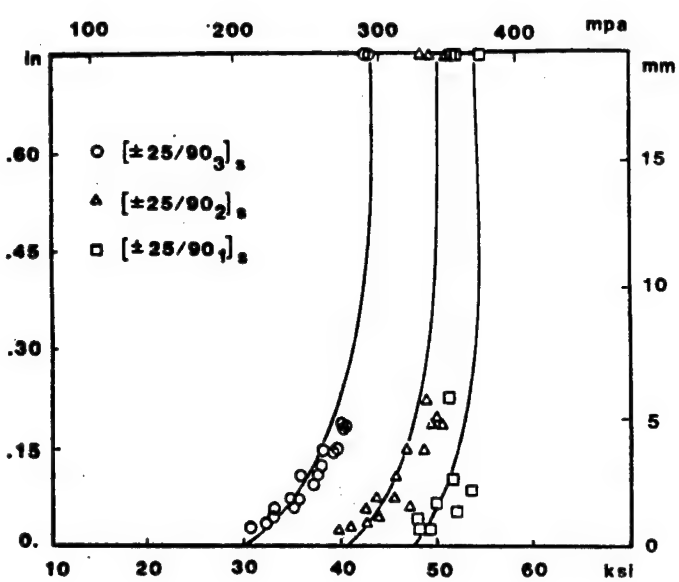


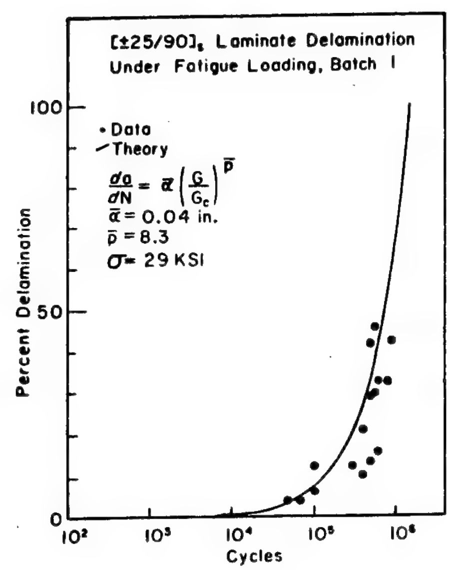


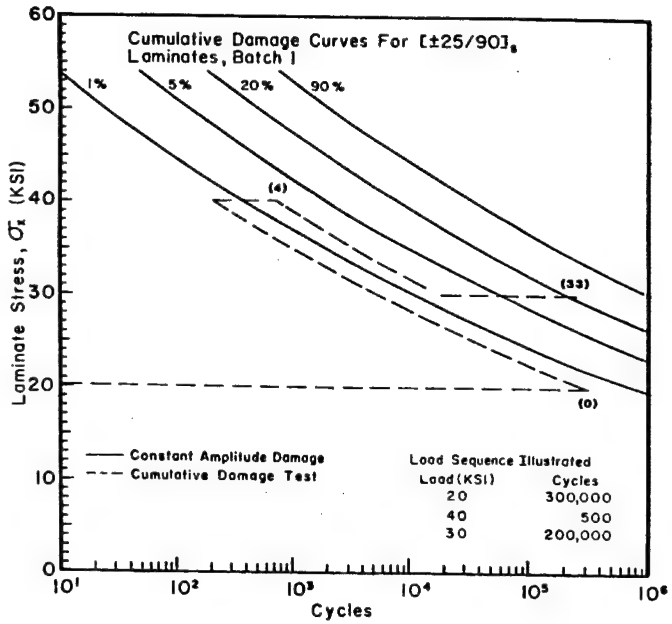
Again, the same ENERGY approach is applied....

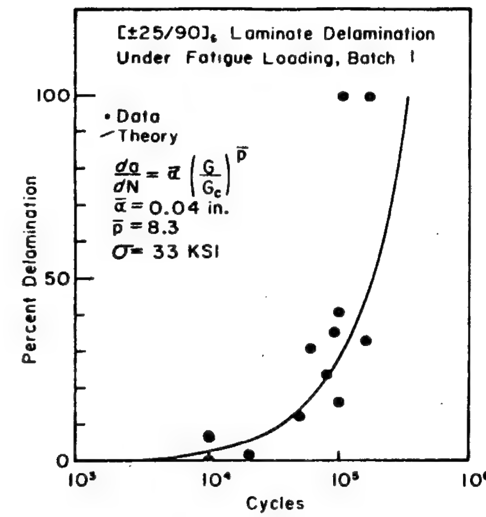


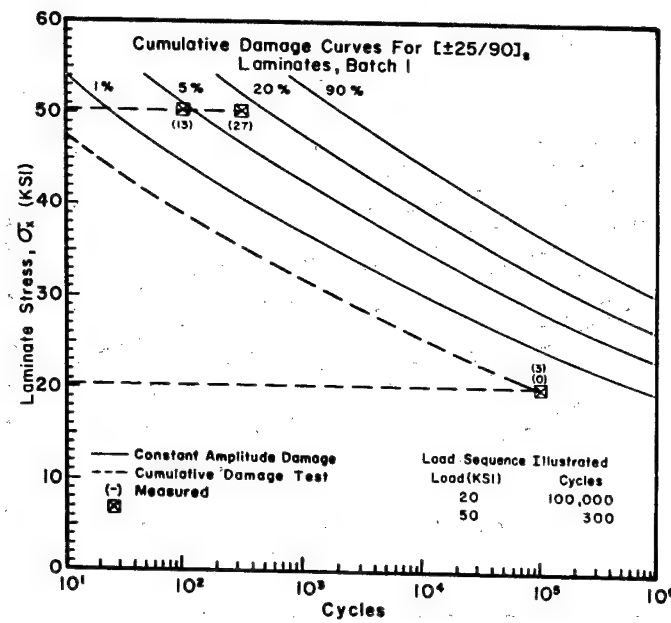
Predicted load range for delamination











CONCLUSIONS

* A model for matrix damage accumulation under time-varying loads is developed.
* 3-D stress fields in multiply cracked laminated media can be calculated.
* In-plane stresses can cause multiple matrix cracks in plies having tensile stress normal to fibers.
* Out-of-plane stresses can cause delamination in interfaces having tensile interlaminar normal stress.
* Intraply and interply cracking modes are generally coupled resulting in multi-ply matrix failures.
* The model simulates these matrix cracks and their growth up to final laminate failure or the breaking of fibers.
* A model for fiber breakage, accounting the presence of matrix cracks, is not included.
* All time-dependent properties of the composite are omitted.

COMPOSITE DEFECT SIGNIFICANCE

S. N. Chatterjee

Materials Sciences Corporation
Gwynedd Plaza II
Bethlehem Pike
Spring House, PA 19477

and

R. B. Pipes and W. A. Dick

Center for Composite Materials
University of Delaware
Newark, DE 19711

ABSTRACT

In recent years various studies have been conducted for nondestructive inspection of different kinds of defects in composite components and assessment of their significance [1-2]. A bibliography on this subject can be found in reference 1. These studies have reached a point where an effort to integrate the technological advances is desirable in order to define a reliable useful methodology for defect evaluations. With this objective, a plan has been formulated for effective utilization of NDE with currently usable inspection methods and criticality assessment techniques. Technology gaps to be filled for immediate practical applications and those which should be addressed for future improvements are identified after a review of inspection methods applicable to each type of defect and criticality criteria for such defects under appropriate kinds of loading. These include, (i) through and part-through flaws like cracks, holes, etc. under in-plane static or fatigue loading; (ii) delaminations near or away from surface flaws and structural discontinuities under in-plane loads; (iii) delaminations under high transverse stresses, and (iii) moisture and porosity under static and fatigue loads.

Analytical studies as well as nondestructive and destructive tests on the following types of AS-3501 laminated composite elements are being conducted to determine the accuracy of criticality assessment techniques.

1. Beam type structures with disbonds under transverse shear - Elasticity solutions are compared with finite element results using constant strain elements. Results from static tests on specimens with multiple disbonds are correlated with analytically predicted failure loads based on total critical energy release rate.

2. Plate type structures with disbonds under transverse stresses - Such structures include laminates with single or multiple elliptic disbonds under transverse shear. Elasticity solutions are compared with approximate plate theory type solutions and test data. Quasistatic failure of single disbonds under combined effects of Mode II and Mode III types of crack opening is considered. Growth of multiple disbonds under combined effects of Mode I, II and III cyclic loading is also being studied. Studies on failure of bonded joints under quasistatic and cyclic tensile loads are being attempted using similar criticality criteria.

3. Plate type structures under in-plane compressive loads - Results from past studies indicate that buckling failure of disbonded laminates under unidirectional compressive loading may be quite critical under certain circumstances. Biaxial loading of disbonded skins of a sandwich construction is being attempted to study failure under quasistatic load and effects of cyclic loading.

REFERENCES

1. Chatterjee, S. N. and Pipes, R. B., "Composite Defect Significance," NADC 80048-80, August 1981.

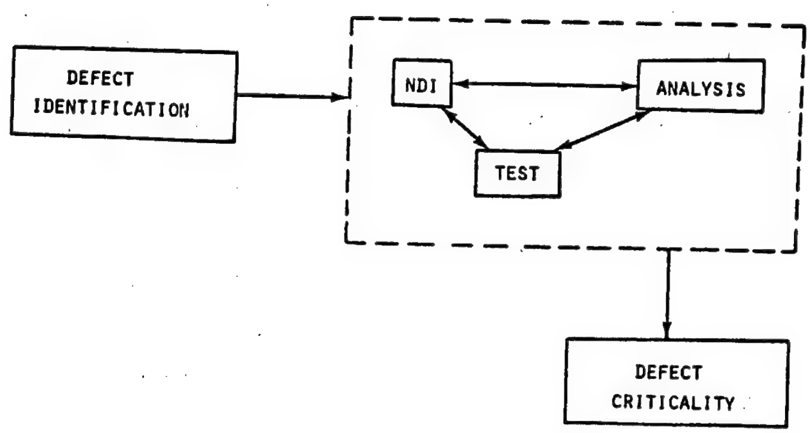
2. "Non-destructive and Analytical Evaluation of Criticality of Defects in Structural Composite Laminates," Symposium Proceedings, NADC under contract no. N62269-80-C-0271, November 1980.

COMPOSITE DEFECT SIGNIFICANCE

WORK PERFORMED UNDER CONTRACTS FROM NADC

SEPTEMBER 1984

COMPOSITE DEFECT SIGNIFICANCE PROGRAM



OBJECTIVE

- DEMONSTRATE THE USE OF NDE METHODOLOGY TO ASSESS FLAW CRITICALITY
  - DEFINE REQUIRED NDI OBSERVATIONS
  - MODEL PHYSICALLY REALISTIC DAMAGE MODES
  - ASSESS PROPERTY DEGRADATION AND FLAW CRITICALITY
  - TRANSLATION OF RESULTS FROM TEST SYSTEMS TO REAL WORLD COMPOSITE STRUCTURES, IDENTIFY TECHNOLOGY GAPS

INSPECTION

TECHNOLOGY GAPS

- STANDARDIZATION NEEDED FOR:
  - PORTABLE ULTRASONICS PULSE-ECHO
  - X-RAY RADIOGRAPHY WITH AND WITHOUT PENETRANTS
  - ULTRASONIC RESONANCE AND SONIC TESTING
  - ULTRASONIC SCANNING FOR QUANTIFYING MOISTURE CONTENT, ETC.
- FURTHER R&D NEEDED FOR:
  - STEROGRAPHIC X-RAY
  - NEUTRON RADIOGRAPHY
  - THERMOGRAPHY
  - HOLOGRAPHY
  - ALTERNATE X-RAY PENETRANTS
  - ACOUSTIC EMISSION
  - MOISTURE & POROSITY MEASUREMENTS
  - TOMOGRAPHY
  - ROBOTICS AND ESAT WITH ULTRASONIC SCANNING
  - ULTRASONIC ATTENUATION AND VELOCITY

CRITICALITY ASSESSMENT

TECHNOLOGY GAPS

- ASSESSMENT REQUIRES INFORMATION ON MAXIMUM STRESS OR LOADS IN COMPONENTS
- NEGLIGIBLE DAMAGE CLASSIFICATION USING
  - STRESS FRACTURE CRITERIA
  - LAMINAE FRACTURE TOUGHNESS
  - APPROXIMATE SUBLAMINATE BUCKLING
  - STATIC DELAMINATION FRACTURE
  - 1-D SELF-SIMILAR DELAMINATION FATIGUE GROWTH
- R&D REQUIRED FOR:
  - TENSILE FATIGUE TESTING
  - F.E. OR RIGOROUS SUBLAMINATE BUCKLING
  - DELAMINATION FRACTURE
    - INTERACTION CRITERION
    - NON SELF-SIMILAR ANALYSES
  - DELAMINATION FATIGUE GROWTH
    - MIXED MODE
    - NON SELF-SIMILAR
  - PART THROUGH DEFECT TESTING
  - MOISTURE & POROSITY TESTING

DISBONDS IN BEAM TYPE STRUCTURES UNDER TRANSVERSE SHEAR

- QUASISTATIC INTERLAMINAR FRACTURE - EFFECTS OF SIZE, LOCATION SHARPNESS DISBOND TIPS AND MULTIPLE DISBONDS

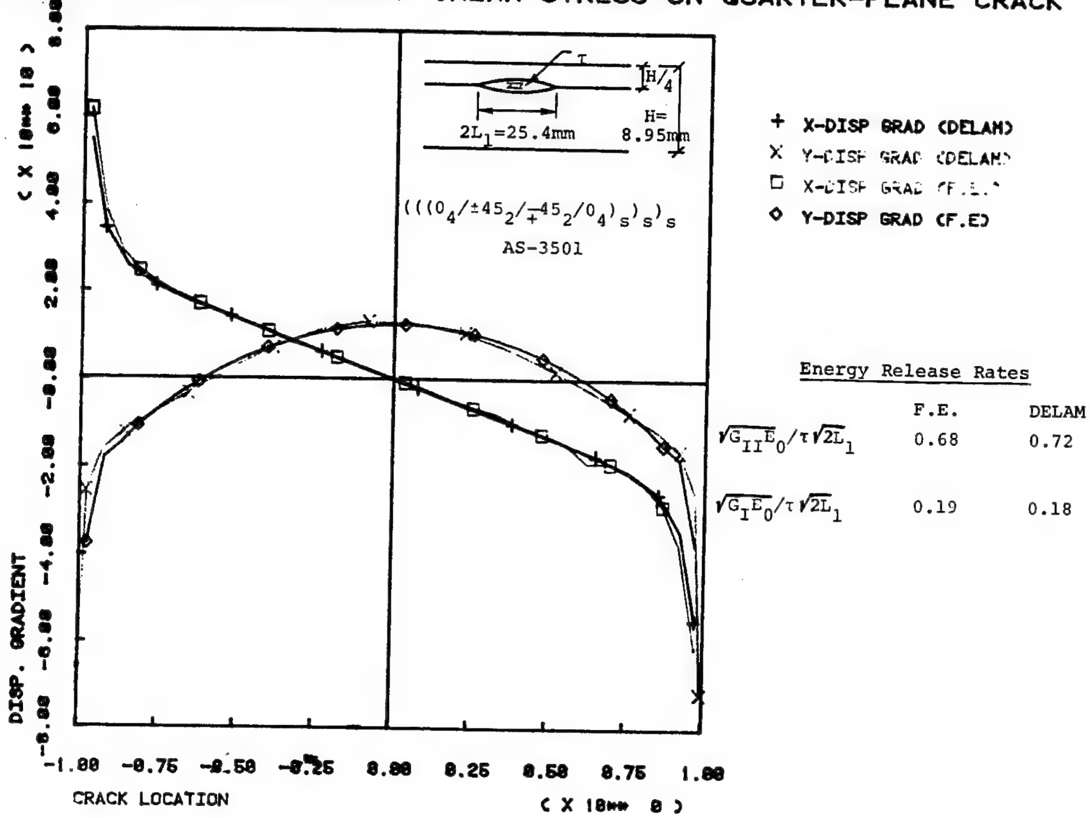
- DISBOND GROWTH UNDER FATIGUE - SEMI-EMPIRICAL GROWTH LAWS

- ULTRASONIC PULSE ECHO TO MONITOR DAMAGE GROWTH

- DIFFERENT ANALYSIS METHODS - COMPARE RESULTS

FINITE ELEMENT AND RIGOROUS ELASTICITY SOLUTIONS

DISPLACEMENT GRAD UNDER SHEAR STRESS ON QUARTER-PLANE CRACK



DISBONDS IN PLATE TYPE STRUCTURES UNDER TRANSVERSE SHEAR

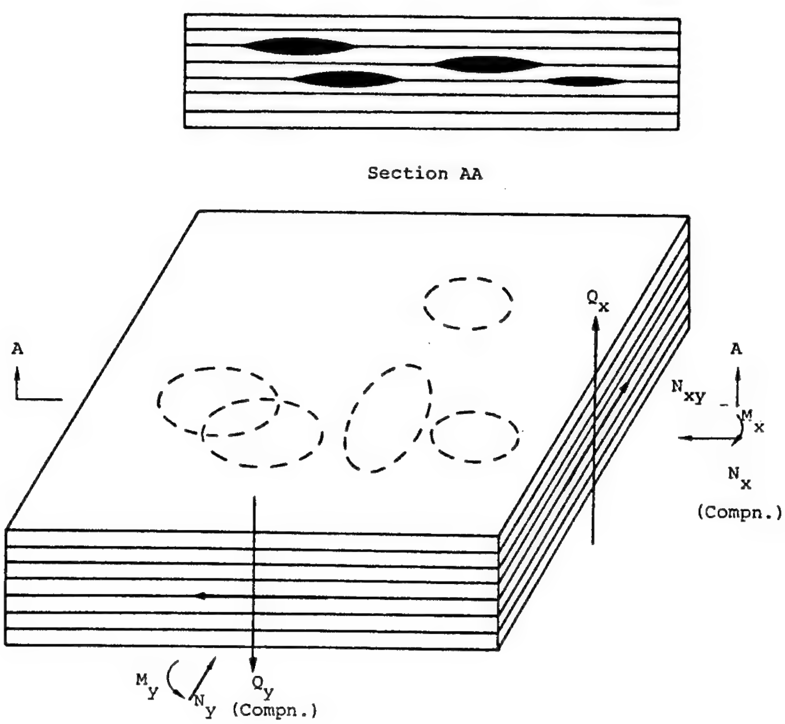
- QUASISTATIC INTERLAMINAR FRACTURE - EFFECTS OF SIZE, SHAPE AND GEOMETRIC DISCONTINUITIES (BONDED JOINTS)

- GROWTH OF SINGLE AND MULTIPLE DISBONDS UNDER FATIGUE - SEMI EMPIRICAL GROWTH LAWS, EFFECTS OF COMPLEX STRESS STATES AND GEOMETRIC DISCONTINUITIES (PLYDROPS AND BONDED JOINTS)

- ULTRASONIC PULSE ECHO SCANNING TO MONITOR DAMAGE GROWTH, GATING TECHNIQUES FOR MULTIPLE DISBONDS

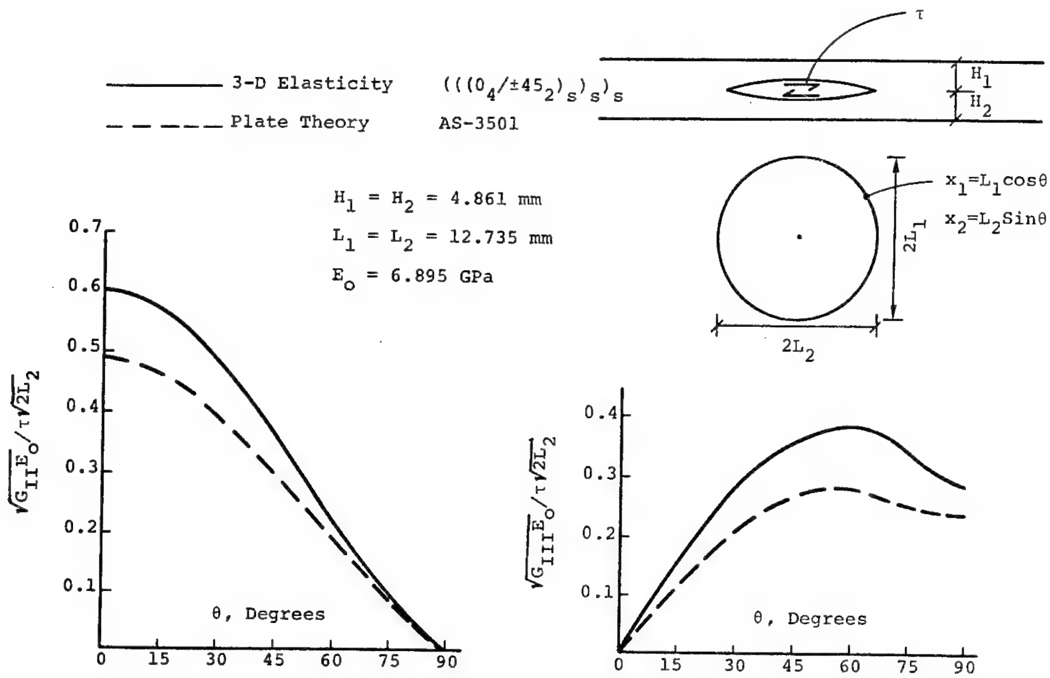
- 3-D ELASTICITY AND PLATE THEORY SOLUTIONS

ELLIPTIC DISBONDS IN A LAMINATED PLATE

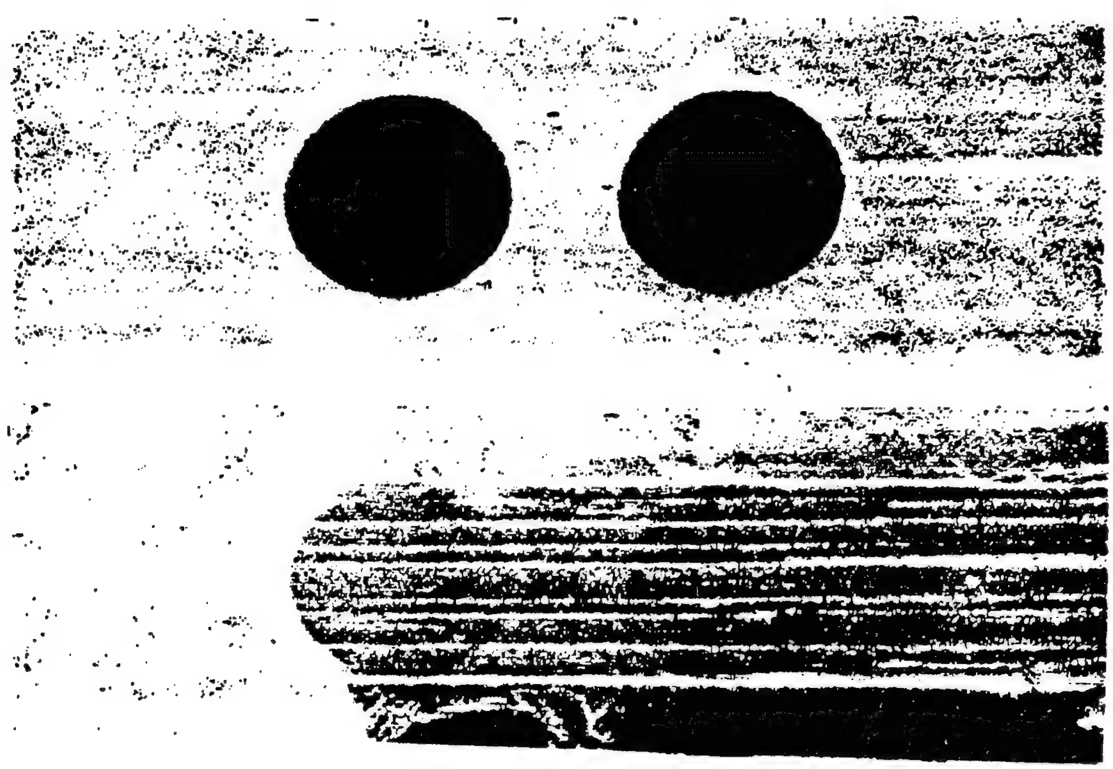


MODE II AND III ENERGY RELEASE RATES

3-D ELASTICITY AND PLATE THEORY SOLUTIONS



QUASISTATIC FAILURE - ELLIPTIC DISBONDS IN THICK PLATES

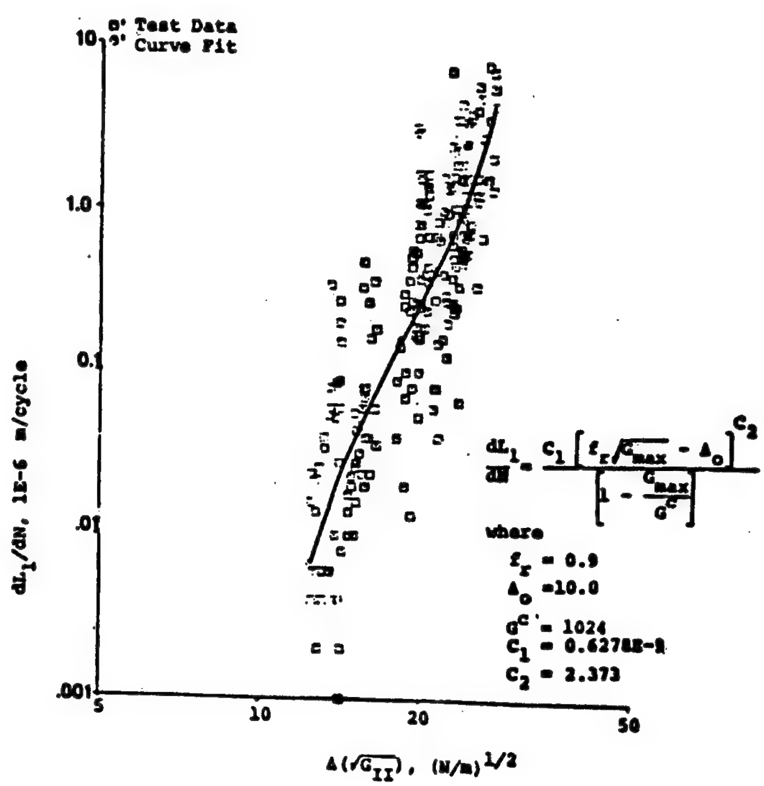


FAILURE OF ELLIPTIC DISBONDS

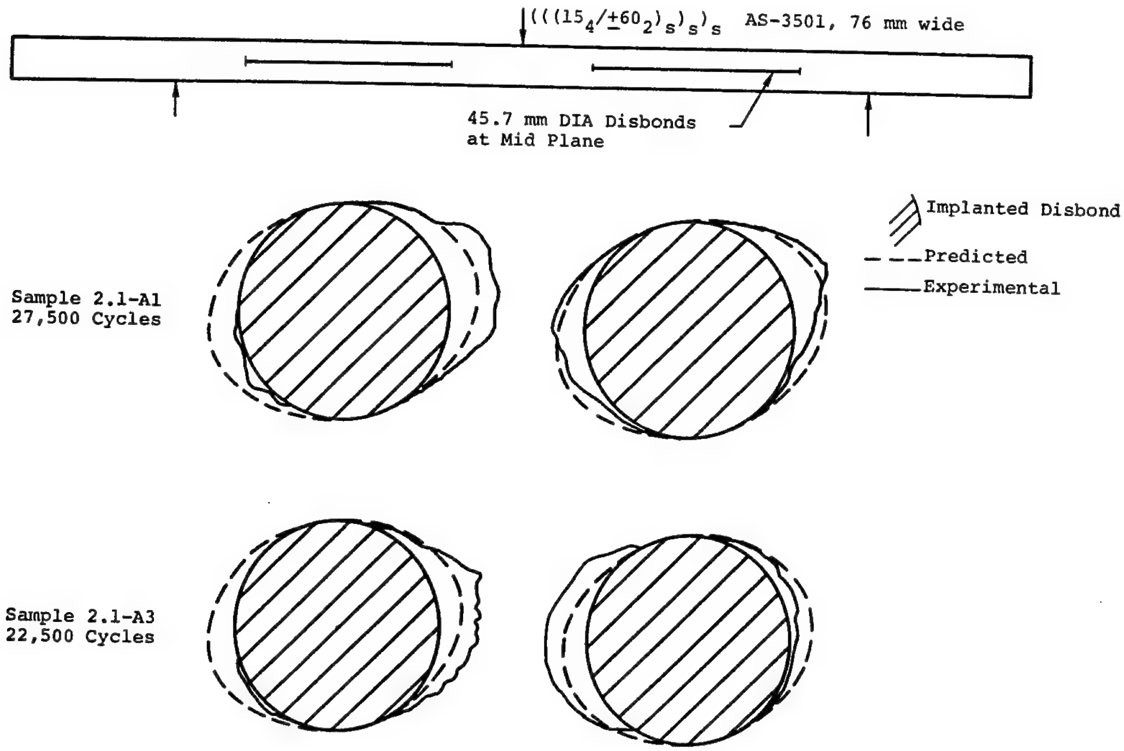
SAMPLE RESULTS

Sample	Point Criterion	Realistic Criterion	Exptl.
B - 2 x 2 - 1	14,800	19,200	17,700
B - 2 x 2 - 2	15,600	20,200	18,300
B - 2 x 2 - 3	15,700	20,400	20,800

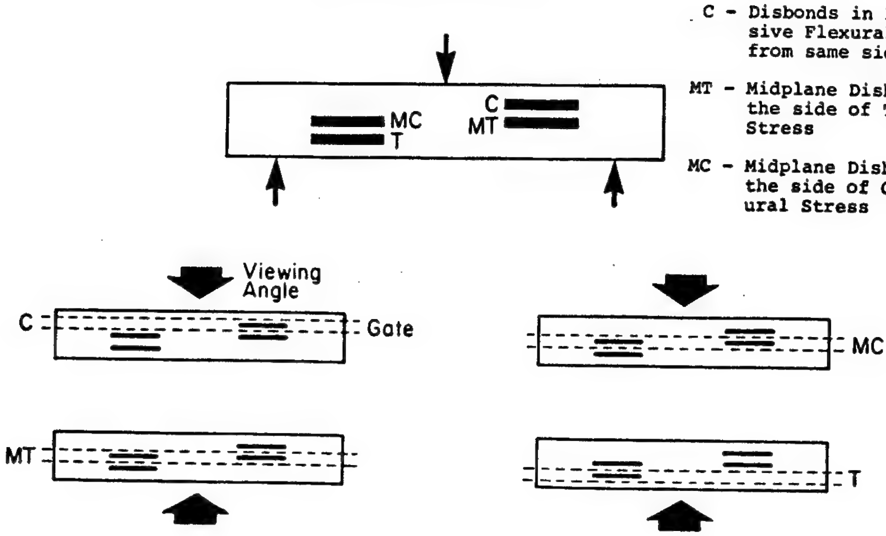
DISBOND GROWTH LAW



DISBOND GROWTH PATTERN



C-SCAN GATING



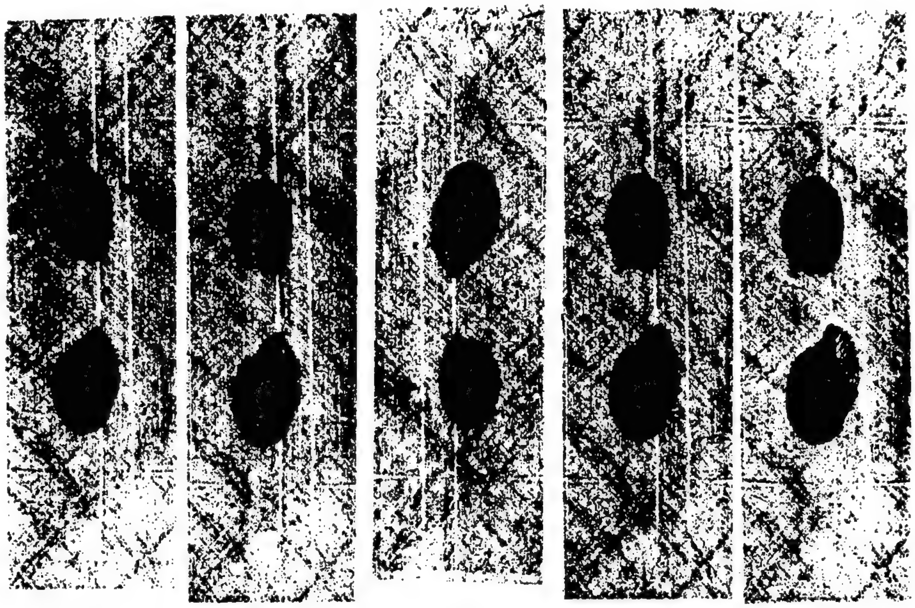
T - Disbond in Region of Tensile Flexural Stress viewed from same side

C - Disbonds in Region of Compressive Flexural Stress viewed from same side

MT - Midplane Disbond viewed from the side of Tensile Flexural Stress

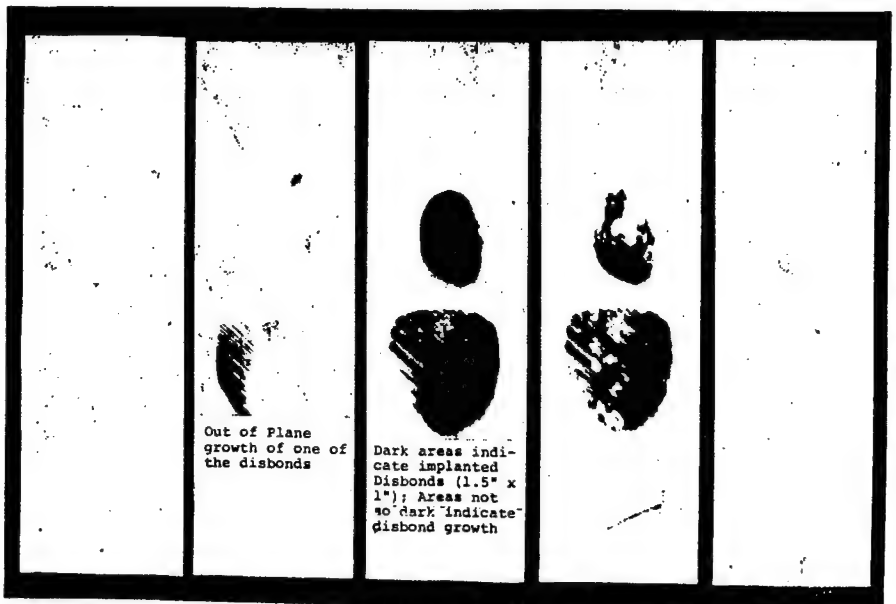
MC - Midplane Disbond viewed from the side of Compressive Flexural Stress

DISBONDS AT -H/4



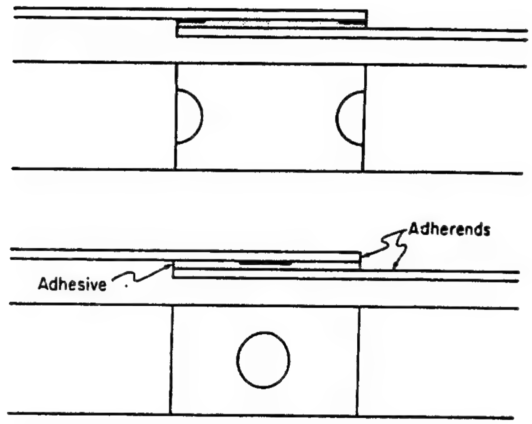
21000 26000 31000 45000 55000

NONCOPLANAR GROWTH OF DISBONDS



above $11^{th}/12^{th}$ ply | above $13^{th}/14^{th}$ ply | at $16^{th}/17^{th}$ ply i.e. Plane of Implanted Disbond | ← below →

BONDED JOINT DISBONDS



FRACTURE SURFACE - ADHEREND ADHESIVE INTERFACE

ADHEREND SURFACE
(160X)

ADHESIVE SURFACE
(160X)

CONCLUSIONS

- TECHNOLOGY IS ADEQUATE FOR NDE. FOR MORE EFFICIENT UTILIZATION FURTHER WORK IS NEEDED FOR
  - STANDARDIZATION OF USABLE NDI
  - DEVELOPMENTS OF NEW NDI
  - NEGLIGIBLE DAMAGE CLASSIFICATIONS
- CRITICALITY ASSESSMENT TECHNIQUES FOR DELAMINATIONS ARE REASONABLY ACCURATE. SIMILAR APPROACH MAY BE APPLICABLE FOR BONDED JOINTS. FOR BETTER UNDERSTANDING, STUDIES ON THE FOLLOWING TOPICS APPEAR NECESSARY.
  - MIXED MODE EFFECTS
  - NON-SELF-SIMILAR AND NONCOPLANAR GROWTH

COMPOSITE MECHANICS/RELATED ACTIVITIES
AT LEWIS RESEARCH CENTER

C. C. CHAMIS

NASA LEWIS RESEARCH CENTER
CLEVELAND, OHIO 44135
(216) 433-4000

ABSTRACT

Lewis research activities and progress in composite mechanics and closely related areas are summarized. The research activities summarized include: (1) Composite Mechanics; (2) Computer Programs for Composites; (3) High Temperature Composites; and (4) Composite Engine Structural Components. The research activity focus is on: (1) Composite Mechanics -- simplified micromechanics equations for strength fracture toughness and impact resistance, finite element substructuring for composite mechanics, laminate analysis and dynamic crack propagation, life/durability and failure modes, and high strain rate effects; (2) Computer Programs for Composites -- integrated composite analyzer (ICAN), structural composite durability, composite thermoviscoplastic structural analysis, and structural tailoring; (3) High Temperature Composites -- test methods development and characterization, composite burner liners, and tungsten-fiber reinforced superalloys (FRS); and (4) Composite Engine Structural Components -- composite ducts, fan blades from superhybrid or with composite in-lays, swept turboprops and props for general aviation aircraft, and FRS turbine blades.

COMPOSITE MECHANICS/RELATED ACTIVITIES
AT LEWIS RESEARCH CENTER

C. C. CHAMIS
NASA LEWIS RESEARCH CENTER
CLEVELAND, OHIO 44135

TENTH ANNUAL MECHANICS OF COMPOSITES REVIEW
DAYTON, OHIO, OCTOBER 15-17, 1984

OBJECTIVE

SUMMARY OF LEWIS RESEARCH ACTIVITIES AND PROGRESS IN:

- o COMPOSITE MECHANICS
- o COMPUTER PROGRAMS FOR COMPOSITES
- o HIGH TEMPERATURE COMPOSITES
- o COMPOSITE ENGINE COMPONENTS

CONCLUSIONS

- o CURRENT LEWIS RESEARCH ACTIVITIES ON COMPOSITE MECHANICS/RELATED AREAS INCLUDE:
  - o COMPOSITE MECHANICS, COMPUTER PROGRAMS FOR COMPOSITES, HIGH TEMPERATURE COMPOSITES AND COMPOSITE ENGINE STRUCTURAL COMPONENTS
  - o RECENT PROGRESS INCLUDES:
    - o SIMPLIFIED MICROMECHANICS EQUATIONS FOR STRENGTH
    - o FINITE ELEMENT SUBSTRUCTURING FOR COMPOSITE MECHANICS (EDGE EFFECT AND DYNAMIC DEFECT PROPAGATION)
    - o APPLICATION OF THE LEWIS LIFE/DURABILITY THEORY
    - o FAILURE MODES OFF-AXIS COMPRESSION, IMPACT AND DYNAMIC DELAMINATION
    - o CONTINUING DEVELOPMENT OF: ICAN, CODSTRAN, N. L. COBSTRAN, STAEBL, STAT
    - o INITIATION OF RESEARCH IN HIGH TEMPERATURE COMPOSITE AND HIGH-STRAIN RATE EFFECTS ON STRESS CONCENTRATION AND ENVIRONMENTAL BEHAVIOR
    - o STRUCTURAL TAILORING OF COMPOSITE FAN BLADES
    - o THERMOVISCOPLASTIC STRUCTURAL ANALYSIS OF TURBINE BLADES MADE FROM TUNGSTEN-FIBER REINFORCED SUPERALLOYS

COMPOSITE MECHANICS

- o SIMPLIFIED MICROMECHANICS EQUATIONS/F. E. VALIDATION
- o SIMPLIFIED MICROMECHANICS EQUATIONS WITH INTERPHASE
- o F. E. SUBSTRUCTURING IN COMPOSITE MECHANICS AND LAMINATE ANALYSIS
- o OFF-AXIS COMPRESSION BEHAVIOR-FAILURE MODES
- o LIFE/DURABILITY IN HYGROTHERMOMECHANICAL ENVIRONMENTS
- o DEVELOPMENT OF HYGROTHERMOMECHANOCHRONIC THEORY
- o DYNAMIC INTERPLY DELAMINATION
- o HIGH-STRAIN-RATE EFFECTS ON STRESS CONCENTRATION AND ENVIRONMENTAL BEHAVIOR
- o DEVELOPMENT OF HYBRID-STRESS FINITE ELEMENTS FOR COMPOSITES WITH CRACKS AND CUTOUTS

COMPUTER PROGRAMS FOR COMPOSITES

o	CRACAN	CRACKED COMPOSITE ANALYZER - USING HYBRID-STRESS FINITE ELEMENTS
o	ICAN	INTEGRATED COMPOSITES ANALYZER
o	COBSTRAN	COMPOSITE BLADE STRUCTURAL ANALYSIS - STAND ALONE
o	CODSTRAN	COMPOSITE DURABILITY STRUCTURAL ANALYSIS
o	N. L. COBSTRAN	NONLINEAR COMPOSITE BLADE STRUCTURAL ANALYSIS
o	STAEBL	STRUCTURAL TAILORING OF ENGINE BLADES
o	STAT	STRUCTURAL TAILORING OF ADVANCED TURBOPROPS

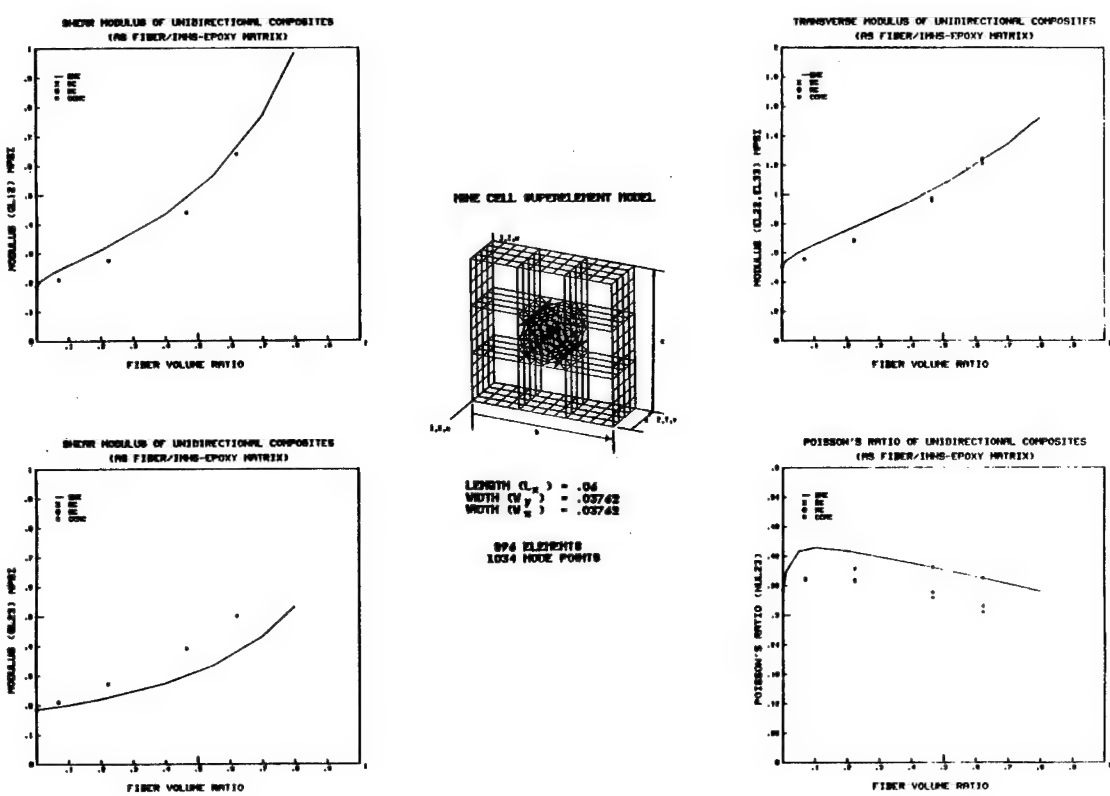
HIGH TEMPERATURE COMPOSITES

- o TEST METHODS AND CHARACTERIZATION
- o COMPOSITE BURNER LINER
- o TUNGSTEN-FIBER REINFORCED SUPERALLOYS

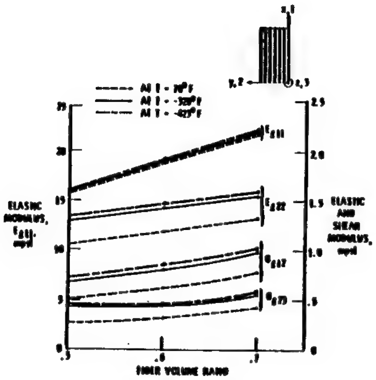
COMPOSITE ENGINE STRUCTURAL COMPONENTS

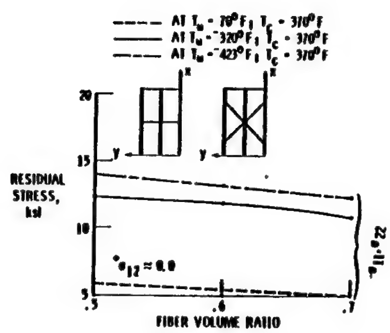
- o COMPOSITE DUCTS
- o FAN BLADES - SUPERHYBRID, COMPOSITE INLAYS
- o SWEPT TURBOPROPS
- o COMPOSITE BLADES FOR GENERAL AVIATION AIRCRAFT ENGINES

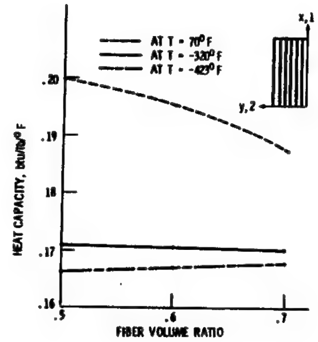
MICROMECHANICS 3-D FINITE ELEMENT CORRELATIONS



COMPOSITE PROPERTIES IN A SPACE ENVIRONMENT







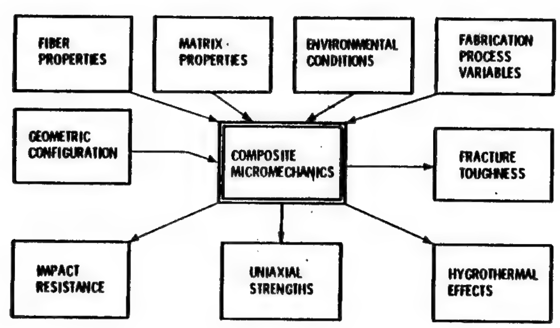
PREDICTED RESULTS VERSUS EXPERIMENTAL DATA

LONGITUDINAL ELASTIC MODULUS, ksi

MATERIAL*	Tu = -300° F		Tu = 70° F		Tu = 200° F	
	ICAN	MEASURED	ICAN	MEASURED	ICAN	MEASURED
COMPOSITE 1	4589	4679	4251	4397	4076	4107
COMPOSITE 2	5587	6643	5395	5964	5457	5981
COMPOSITE 3	4440	5300	4114	4300	3948	4200

* 1 7781 E-GLASS CLOTH
2 7576 E-GLASS CLOTH
3 REPRESENTATIVE LAMINATE; COMBINATION OF 7781 AND 7576 GLASS

SIMPLIFIED COMPOSITE MICROMECHANICS EQUATIONS FOR STRENGTH



COMPOSITE MICROMECHANICS: UNIAXIAL STRENGTHS - IN-PLANE

1. LONGITUDINAL TENSION: $S_{\ell 11T} \approx k_f S_{fT}$
2. LONGITUDINAL COMPRESSION:
   - FIBER COMPRESSION: $S_{\ell 11C} \approx k_f S_{fC}$
   - DELAMINATION/SHEAR: $S_{\ell 11C} \approx 10 S_{\ell 12S} + 2.5 S_{mT}$
   - MICROBUCKLING: $S_{\ell 11C} \approx \dfrac{G_m}{1 - k_f\left(1 - \dfrac{G_m}{G_{f12}}\right)}$
3. TRANSVERSE TENSION: $S_{\ell 22T} \approx \left[1 - (\sqrt{k_f} - k_f)(1 - E_m/E_{f22})\right] S_{mT}$
4. TRANSVERSE COMPRESSION: $S_{\ell 22C} \approx \left[1 - (\sqrt{k_f} - k_f)(1 - E_m/E_{f22})\right] S_{mC}$
5. INTRALAMINAR SHEAR: $S_{\ell 12S} \approx \left[1 - (\sqrt{k_f} - k_f)(1 - G_m/G_{f12})\right] S_{mS}$
6. FOR VOIDS: $S_m \approx \left\{1 - \left[4k_v/(1 - k_f)\pi\right]^{1/2}\right\} S_m$

COMPOSITE MICROMECHANICS: UNIAXIAL STRENGTHS - THROUGH-THE-THICKNESS

1. INTERLAMINAR SHEAR: $S_{\ell 13S} \approx \left[1 - (\sqrt{k_f} - k_f)(1 - G_m/G_{f12})\right] S_{mS}$

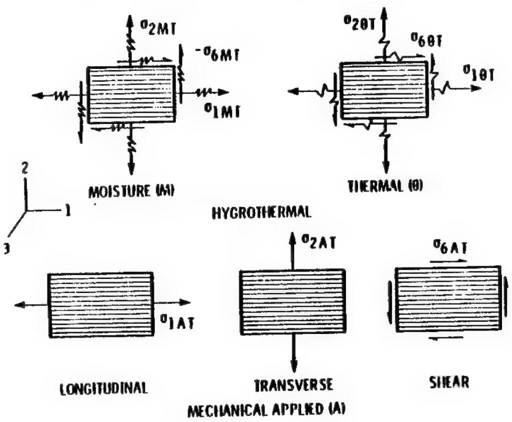
   $S_{\ell 23S} \approx \left[\dfrac{1 - \sqrt{k_f}\,(1 - G_m/G_{f23})}{1 - k_f\,(1 - G_m/G_{f23})}\right] S_{mS}$
2. SHORT-BEAM-SHEAR: $S_{\ell 13SB} \approx 1.5\, S_{\ell 13S}$

   $S_{\ell 23SB} \approx 1.5\, S_{\ell 23S}$
3. FLEXURAL: $S_{\ell 11F} \approx \dfrac{3 k_f S_{fT}}{1 + \dfrac{S_{fT}}{S_{fC}}}$

   $S_{\ell 22F} \approx \dfrac{3\left[1 - (\sqrt{k_f} - k_f)(1 - E_m/E_{f22})\right] S_{mT}}{1 + \dfrac{S_{mT}}{S_{mC}}}$
4. FOR VOIDS: $S_m \approx \left\{1 - \left[4k_v/(1 - k_f)\pi\right]^{1/2}\right\} S_m$

FAILURE CRITERIA WITH "SENSE PARITY"

HYGROTHERMOMECHANICAL STRESS STATES



ELEMENTARY THEORIES WITH SENSE-PARITY

WORK TO FRACTURE

$$\left(\frac{\sigma_{lA\alpha}}{S_{l\alpha}}\right)^2 + \left(\frac{\sigma_{lM\beta}}{S_{l\beta}}\right)^2 + \left(\frac{\sigma_{l\theta\gamma}}{S_{l\gamma}}\right)^2 + 2\left[\frac{\sigma_{lA\alpha}\,\sigma_{lM\beta}}{S_{l\alpha}\,S_{l\beta}} + \frac{\sigma_{lA\alpha}\,\sigma_{l\theta\gamma}}{S_{l\alpha}\,S_{l\gamma}} + \frac{\sigma_{lM\beta}\,\sigma_{l\theta\gamma}}{S_{l\beta}\,S_{l\gamma}}\right] \leq 1$$

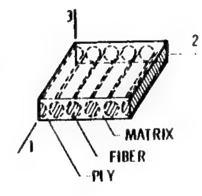
NOTATION:

l = 1, 2, 6 = LONGITUDINAL, TRANSVERSE, SHEAR

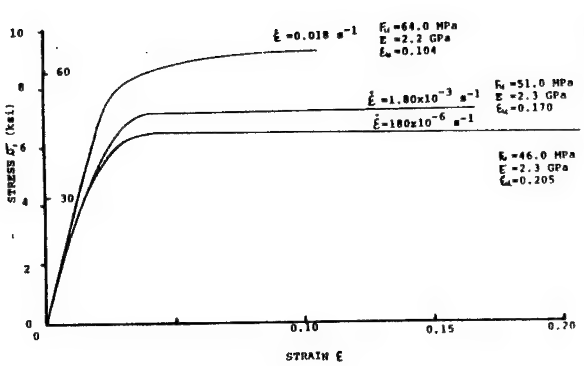
α, β, γ = T OR C = TENSION OR COMPRESSION

σ = STRESS

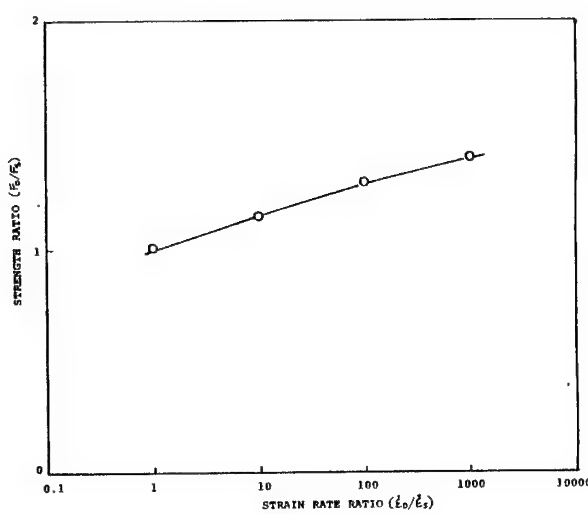
S = STRENGTH (FRACTURE STRESS)



HIGH STRAIN RATE EFFECTS ON RESIN AND COMPOSITE PROPERTIES

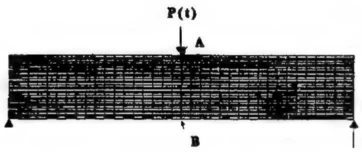


F-185 resin under tensile loading

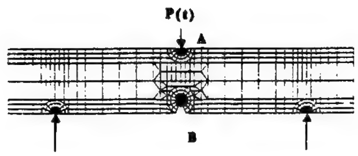


Tensile strength ratio of F-185/T300 90-deg unidirectional composite

STRESS WAVE PROPAGATIONS IN SMOOTH AND NOTCHED SPECIMENS

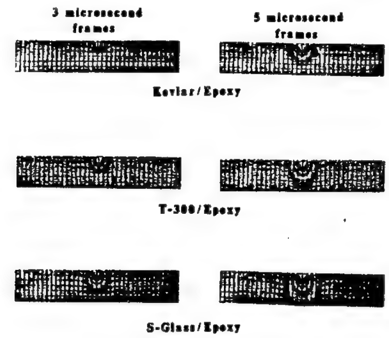


Smooth Specimen

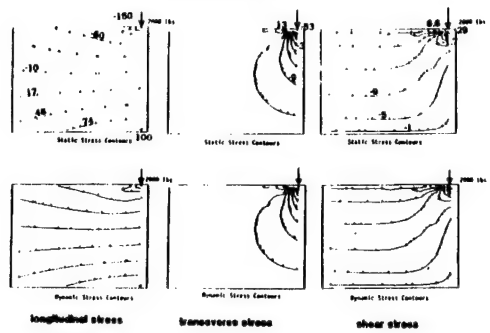


Notched Specimen

DISPLACEMENT PROPAGATION

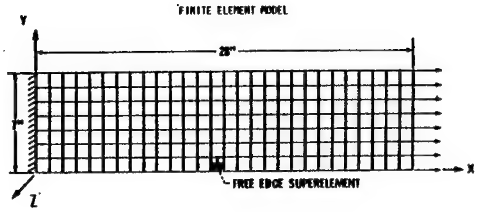


DYNAMIC AND STATIC STRESS CONTOURS FOR SMOOTH S-GLASS/EPOXY SPECIMEN

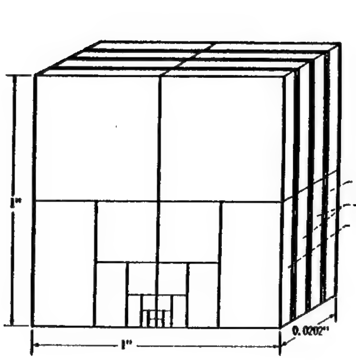


EDGE STRESSES VIA FINITE ELEMENT SUBSTRUCTURING

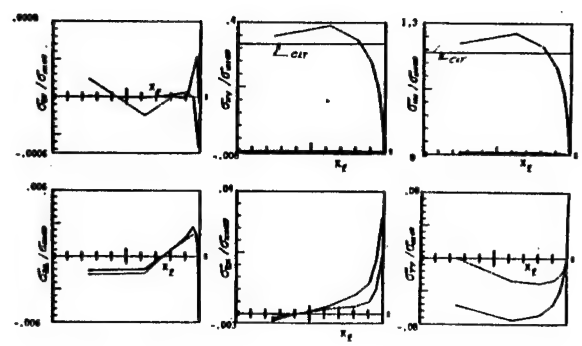
FINITE ELEMENT MODEL



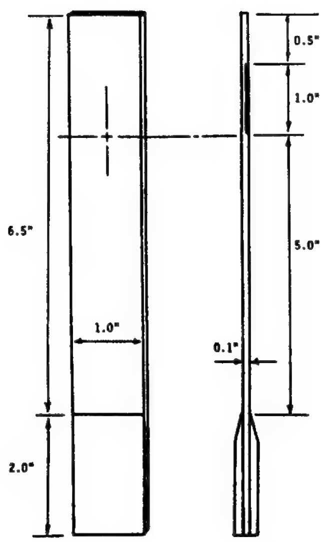
FREE EDGE SUPERELEMENT



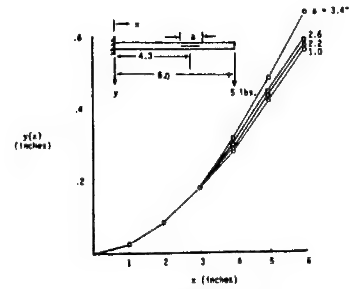
3-D PLY AND INTERPLY STRESS FIELDS AS THE FREE EDGE IS APPROACHED (+20° PLY, $(\pm 20)_s$ AS/E LAMINATE)



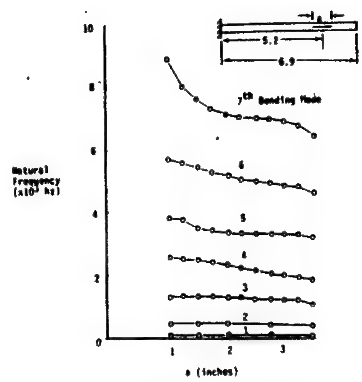
DELAMINATION EFFECTS ON STIFFNESS AND FREQUENCY



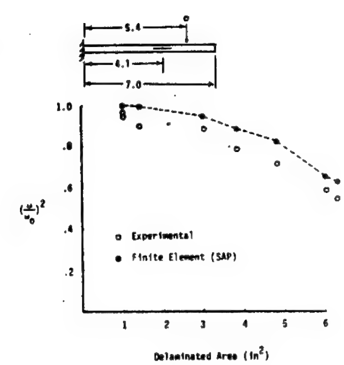
IMPACT SPECIMEN GEOMETRY



STIFFNESS (F.E. PREDICTION)

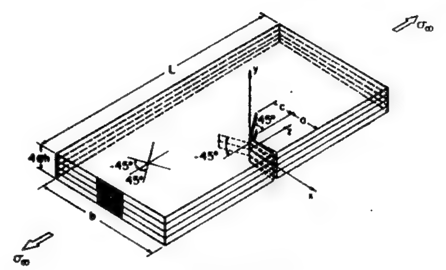


FREQUENCY (F.E. PREDICTION)

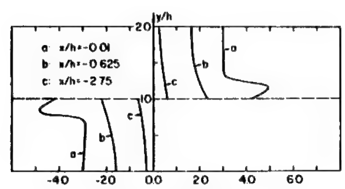


FREQUENCY

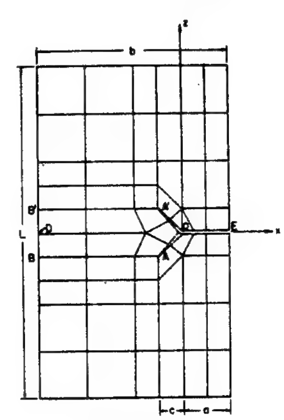
3-D STRESS FIELDS IN INTERNAL DEFECTS



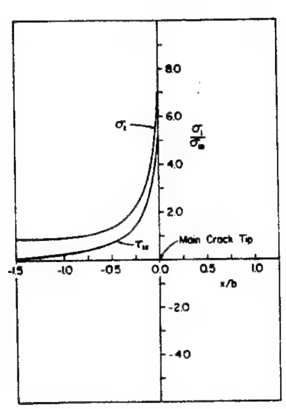
SPECIMEN GEOMETRY $(\pm45)_s$



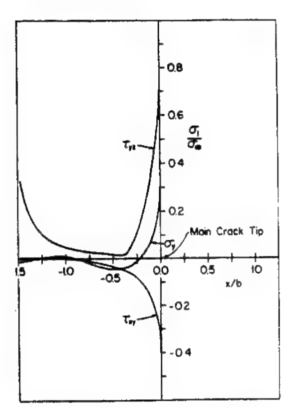
IN-PLANE SHEAR STRESS (Z=0)



CRACK-BRANCH MODELING

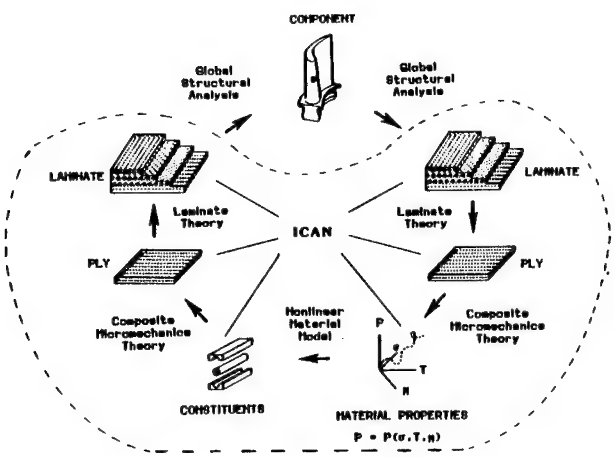


IN-PLANE STRESS

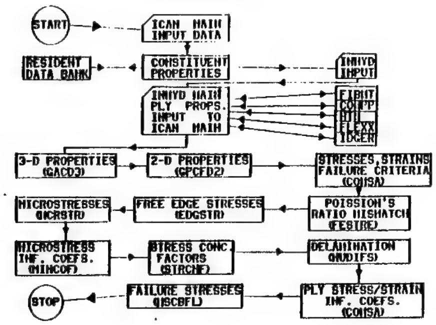


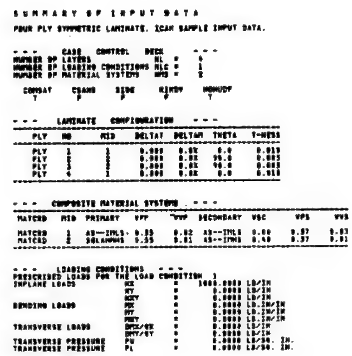
INTERLAMINAR STRESSES (+45 PLY, Z=0)

ICAN: INTEGRATED COMPOSITES ANALYZER



ICAN FLOW CHART





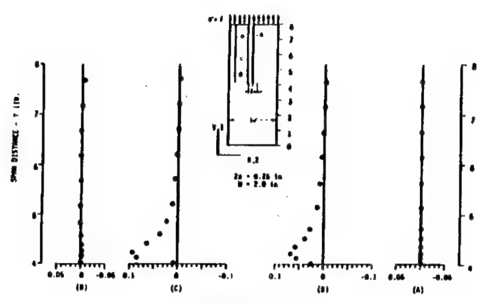
PROGRESSIVE FRACTURE/FRACTURE SURFACE CHARACTERISTICS

CODSTRAN PREDICTED FRACTURE MODES

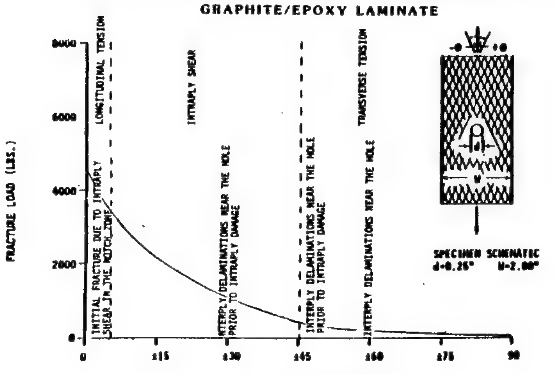
NOTCH TYPE	PLY ORIENTATION; $[\pm\theta]_s$; θ IN DEGREES*									
	0	3	5	10	15	30	45	60	75	90
UNNOTCHED--SOLID	LT	LT, S^3	LT, S^3	LT, S^3	I S	S	I S	TT	TT	TT
NOTCHED--THRU SLIT	S^1 LT	S^1 LT	S^1 LT	S	S	I^4 S	I^4 S	I^4 TT S^2	TT	TT
NOTCHED--THRU HOLE	S^1 LT	S^1 LT	S^1 LT	S	S LT	I^4 S	I^4 S TT	I^4 TT	TT	TT

* LT = LONGITUDINAL TENSION
TT = TRANSVERSE TENSION
S = INTRAPLY SHEAR:
1) INITIAL FRACTURE DUE TO INTRAPLY SHEAR IN THE NOTCH TIP ZONE
2) MINIMAL INTRAPLY SHEARING DURING FRACTURE
3) SOME INTRAPLY SHEAR OCCURRING NEAR CONSTRAINTS (GRIPS)
4) DELAMINATIONS OCCUR IN NOTCH TIP ZONE PRIOR TO ANY INTRAPLY DAMAGE
I = INTERPLY DELAMINATION

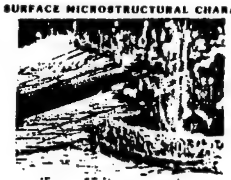
PLY INTRALAMINAR SHEAR STRESS MAGNITUDE PROFILES IN THE UNIDIRECTIONAL GRAPHITE/EPOXY NOTCH/SLIT COMPOSITE



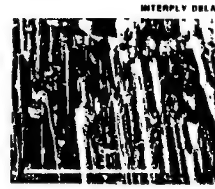
PREDOMINANT FRACTURE MODES FOR THE NOTCH/HOLE GRAPHITE/EPOXY LAMINATE



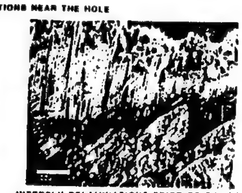
FRACTURE SURFACE MICROSTRUCTURAL CHARACTERISTICS



INTERPLY DELAMINATIONS NEAR THE HOLE

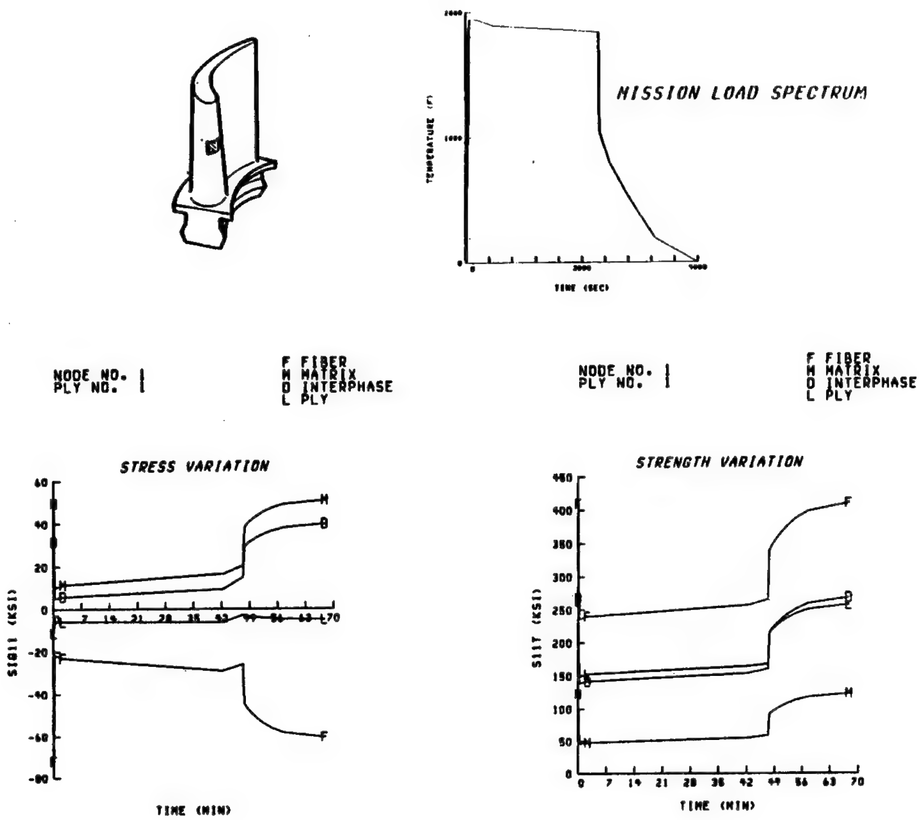


INTRAPLY SHEAR

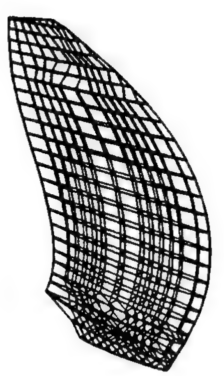


INTERPLY DELAMINATIONS PRIOR TO DAMAGE

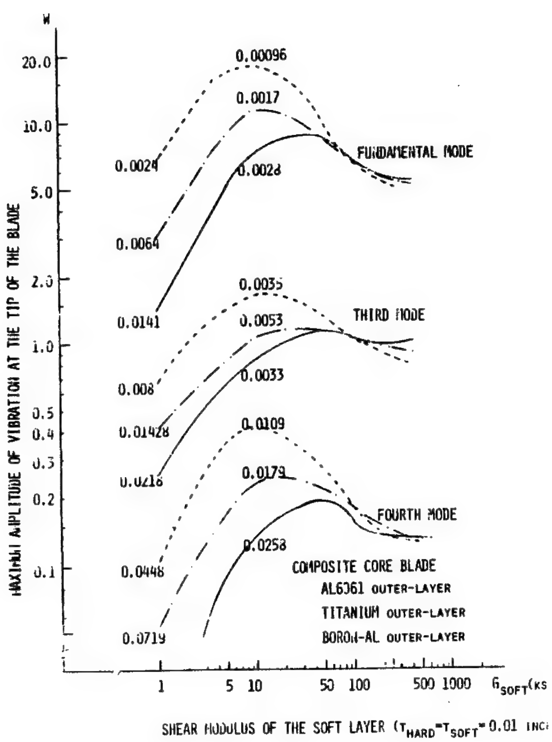
NONLINEAR COBSTRAN



CONSTRAINED LAYER DAMPING OF ADVANCED PROPFANS



VIBRATION MODE
OF A TURBINE BLADE



NONLINEAR DYNAMIC RESPONSE OF COMPOSITE ROTOR BLADES

JOHN J. ENGBLOM AND OZDEN O. OCHOA

DEPARTMENT OF MECHANICAL ENGINEERING
TEXAS A&M UNIVERSITY
COLLEGE STATION, TX 77843

ABSTRACT

Fundamental to this work is the development of a continuum formulation that can accurately account for the effects of interlaminar shear and interlaminar normal stress variation thru-the-thickness of a laminate. Furthermore, emphasis is particularly on tapered-twisted airfoil geometries which can be analytically represented as an assemblage of thin to moderately thick finite elements. To achieve solution efficiencies, the elements developed in this work are of the triangular/quadrilateral plate type as opposed to solid type elements.

On the basis of these requirements and considering viable alternatives, two suitable continuum formulations have been developed and incorporated within a finite element framework. These are herein denoted as the (i) Higher Order Displacement, and (ii) Modified-Kirchhoff formulations, respectively. A computer code has been developed to test the various elements on the basis of small displacement correlations with known analytical and numerical solutions. It is noted that the code has some unique features, e.g., it can assemble elements having an unequal number of degrees of freedom at its nodes, it treats arbitrary ply orientations and it performs integration on a layer-by-layer basis through the laminate.

Significant efforts have also been devoted to developing a suitable large displacement formulation. Due to the requirement that interlaminar stresses be accurately represented, a total Lagrangian formulation is utilized and is based upon the complete Green's strain tensor. A geometric and large-displacement stiffness formulation has been implemented in the computer code based upon a form of the nonlinear strain-nodal displacement relationships suitable for each of the elements under development.

Since emphasis in this work is on the development of incremental response solutions, the computational approach must have the capability to (i) predict and differentiate between relevant failure modes, (ii) modify constitutive equations appropriately and (iii) perform equilibrium iterations to assure stress redistribution based upon the extent of damage. Use of "piecewise smooth" failure criteria in conjunction with "damage state" variables provide a good basis for incrementally tracking damage. This approach is currently being incorporated in the computer code.

NONLINEAR DYNAMIC RESPONSE OF COMPOSITE ROTOR BLADES

JOHN J. ENGBLOM AND OZDEN O. OCHOA

DEPARTMENT OF MECHANICAL ENGINEERING
TEXAS A&M UNIVERSITY
COLLEGE STATION, TX 77843

OBJECTIVES

DEVELOP NONLINEAR DISPLACEMENT/DAMAGE MODELS TO PREDICT STRUCTURAL IMPACT RESPONSE OF COMPOSITE ROTOR BLADES

INCORPORATING

- TRANSVERSE DEFORMATION AND THROUGH-THE-THICKNESS EFFECTS
- APPROPRIATE DAMAGE CRITERIA ON A LAYER-BY-LAYER BASIS

TASKS

TASK I. NONLINEAR DISPLACEMENT FORMULATION FOR COMPOSITE PLATES

- ELEMENT FORMULATION
- LARGE DISPLACEMENT FORMULATION
- INCREMENTAL EQUILIBRIUM FORMULATION

TASK II. INCORPORATE DAMAGE MECHANISM INTO DYNAMIC RESPONSE FORMULATION

TASK III. CORRELATION OF FORMULATED RESPONSE MODEL WITH EXPERIMENTAL DATA

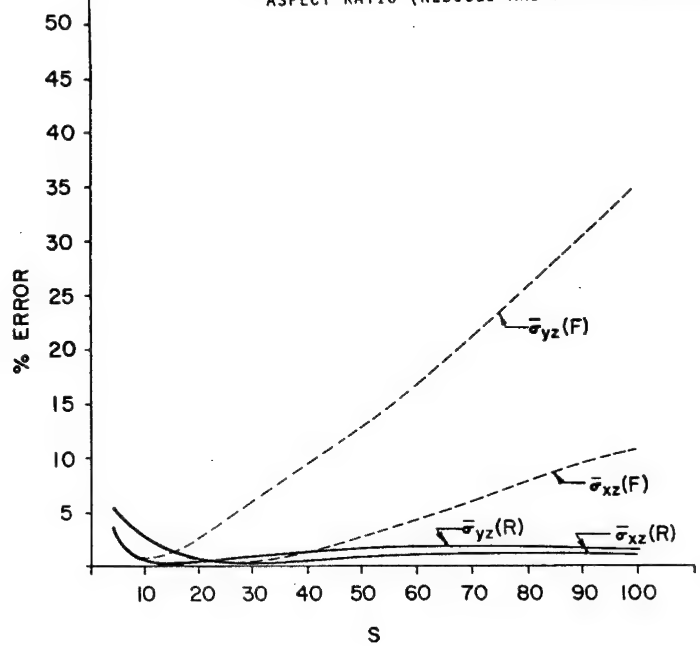
ELEMENT FORMULATION

		HIGHER ORDER DISPLACEMENT MODEL QHD 40	MODIFIED- KIRCHHOFF MODEL QD 32
DEGREES OF FREEDOM	CORNER NODES	$u_o \quad v_o \quad w_o \quad \psi_x \quad \psi_y \quad \phi_x \quad \phi_y$	$u_o \quad v_o \quad w \quad \frac{\partial w}{\partial x} \quad \frac{\partial w}{\partial y} \quad \gamma_x \quad \gamma_y$
	MID-SIDE NODES	$w_o \ \psi_x \ \psi_y$	$w \quad \frac{\partial w}{\partial n}$
DISPLACEMENT FIELD		$u = u_o + z\,\psi_x + z^2\,\phi_x$ $v = v_o + z\,\psi_y + z^2\,\phi_y$ $w = w_o$	$u = u_o - (z\,\frac{\partial w}{\partial x} + \gamma_x)$ $v = v_o - (z\,\frac{\partial w}{\partial y} + \gamma_y)$ $w = [1\ x\ \ y\ x^2\ xy\ y^2\ x^3\ x^2y\ xy^2\ y^3\ x^4\ x^3y\ xy^3\ y^4\ x^4y\ xy^4]\ \{\alpha\}$
STRESS FIELD		CONSTITUTIVE EQS: $\sigma_{xx} = f\ (z^2,\ x^2,\ y^2)$ $\sigma_{yy} = f\ (z^2,\ x^2,\ y^2)$ $\sigma_{xy} = f\ (z^2,\ x^2,\ y^2)$ EQUILIBRIUM EQS: $\sigma_{xz} = f\ (z^3,\ x,\ y)$ $\sigma_{yz} = f\ (z^3,\ x,\ y)$ $\sigma_{zz} = f\ (z^3)$	CONSTITUTIVE EQS: $\sigma_{xx} = f\ (z,\ x^2,\ y^2)$ $\sigma_{yy} = f\ (z,\ x^2,\ y^2)$ $\sigma_{xy} = f\ (z,\ x^2,\ y^2)$ EQUILIBRIUM EQS: $\sigma_{xz} = f\ (z^2,\ x^2,\ y^2)$ $\sigma_{yz} = f\ (z^2,\ x^2,\ y^2)$ $\sigma_{zz} = f\ (z^2,\ x,\ y)$

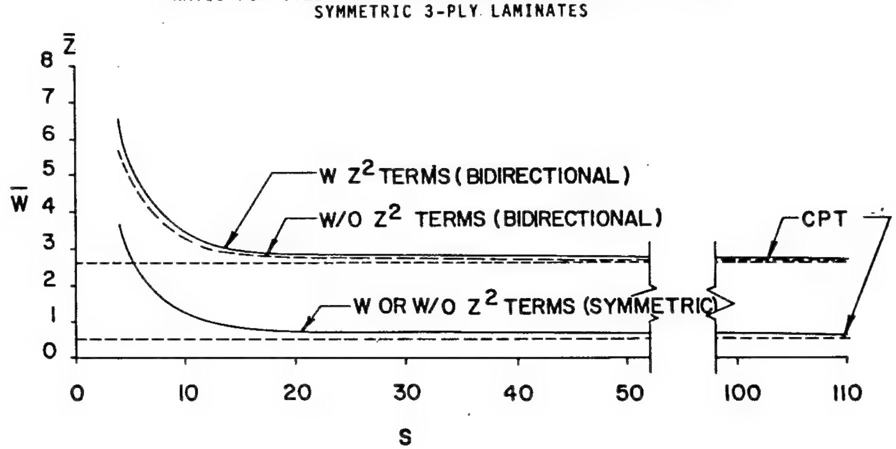
SELECTED DAMAGE CRITERIA

	FIBER FAILURE		MATRIX FAILURE		DELAMINATION
	TENSILE	COMPRESSIVE	TENSILE	COMPRESSIVE	
HASHIN	$\left(\frac{\sigma_{11}}{\sigma_A}\right)^2 + \left(\frac{\sigma_{12}^2 + \sigma_{13}^2}{\tau_A^2}\right) = 1$ or $\sigma_{11} = \sigma_A$	----	$\left(\frac{\sigma_{22} + \sigma_{33}}{\sigma_T^2}\right)^2 +$ $\frac{\sigma_{23}^2 - \sigma_{22}\sigma_{33}}{\tau_T^2} +$ $\frac{\sigma_{12}^2 + \sigma_{13}^2}{\tau_A^2} = 1$	$\frac{1}{\sigma_T}\left(\left(\frac{\sigma_T}{2\tau_T}\right)^2 - 1\right)(\sigma_{22} + \sigma_{33})$ $+ \frac{1}{4\tau_T^2}\ (\sigma_{22} + \sigma_{33})^2 +$ $\frac{1}{\tau_T^2}\ (\sigma_{23}^2 - \sigma_{22}\sigma_{33}) +$ $\frac{1}{\tau_T^2}\ (\sigma_{12}^2 + \sigma_{13}^2) = 1$	----
LEE	$\sigma_{11} = \sigma_A$ $\sigma_{12}^2 + \sigma_{12}^2 = \tau_A^2$	----	$\sigma_{22} = \sigma_T$ $\sigma_{12}^2 + \sigma_{23}^2 = \tau_T^2$	----	$\frac{\sigma_{33}}{\sigma} = 1$ $\frac{\sigma_{13}^2 + \sigma_{23}^2}{\tau^2} = 1$
GRESZCZUK	----	(Buckling) $\sigma_c = \frac{G_r}{1-k}$	----	----	----
	----	----	----	----	$\left(\frac{\sigma_{33}}{\sigma}\right)^2 + \left(\frac{\sigma_{13}^2 + \sigma_{23}^2}{\tau^2}\right) = 1$

ERROR IN NORMALIZED TRANSVERSE SHEAR STRESSES VS. ASPECT RATIO (REDUCED AND FULL INTEGRATION)

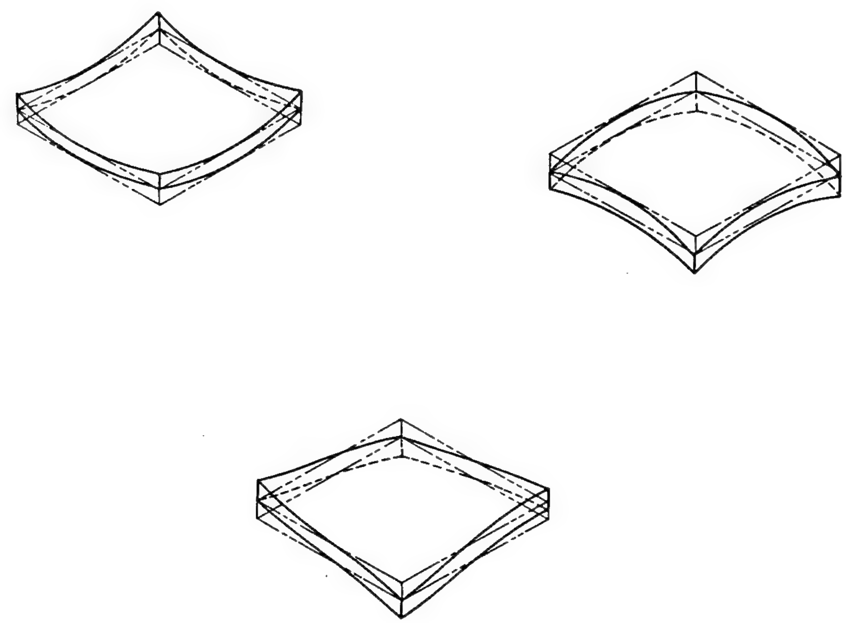


NORMALIZED TRANSVERSE DISPLACEMENT VS. ASPECT RATIO FOR CYLINDRICAL BENDING OF BIDIRECTIONAL AND SYMMETRIC 3-PLY LAMINATES

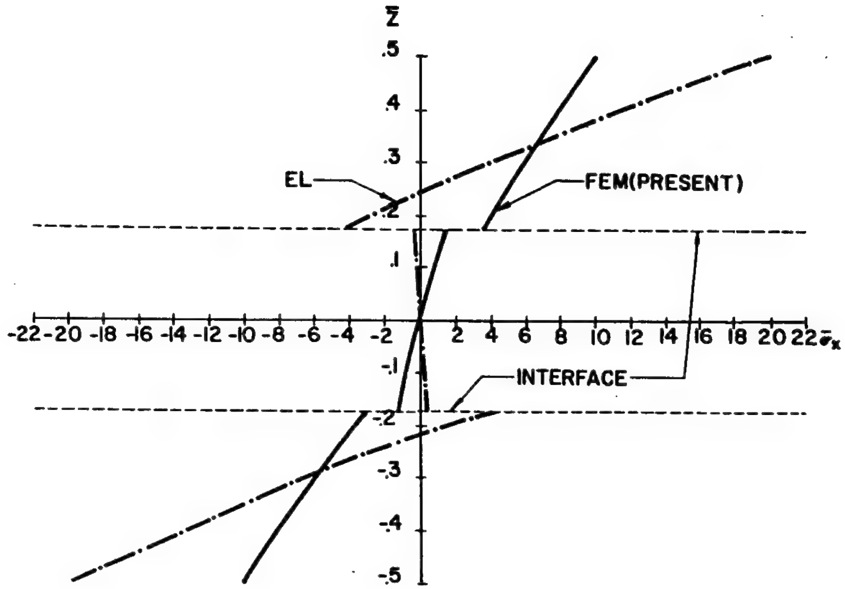


SPURIOUS ZERO ENERGY MODES FOR THE QHD ELEMENTS

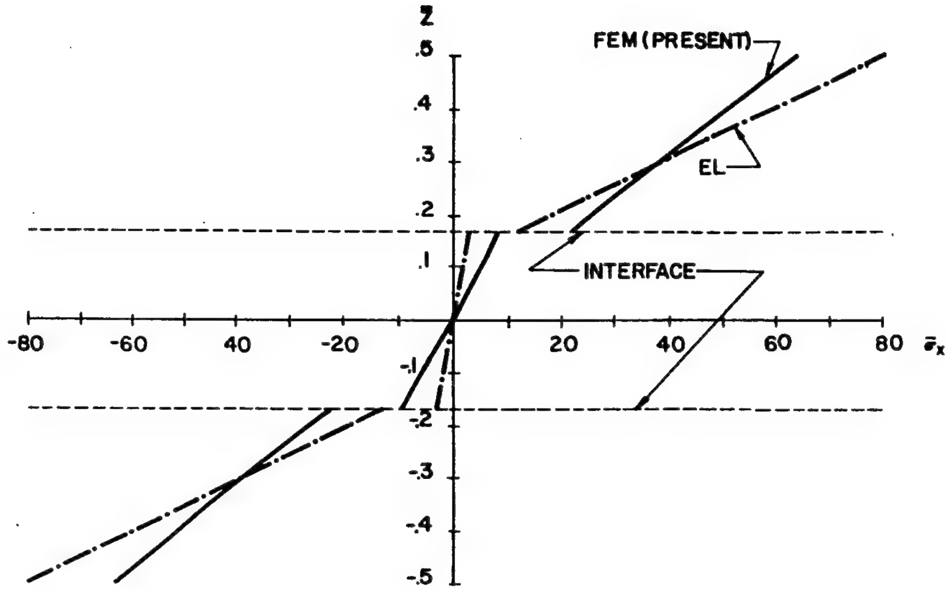
ELEMENT	QUADRATURE ORDER	NUMBER OF ZERO EIGENVALUES	NUMBER OF SPURIOUS MODES
QHD40	3x3 with 2x2 for transverse shear terms	6	0
QHD28	2x2 with 1x1 for transverse shear terms	9	3
QHD20	2x2 with 1x1 for transverse shear terms	8	2



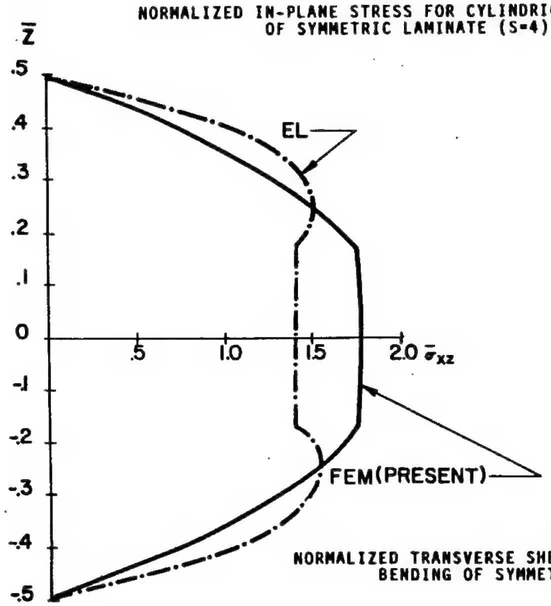
SPURIOUS ZERO ENERGY MODES (QHD28 ELEMENT)



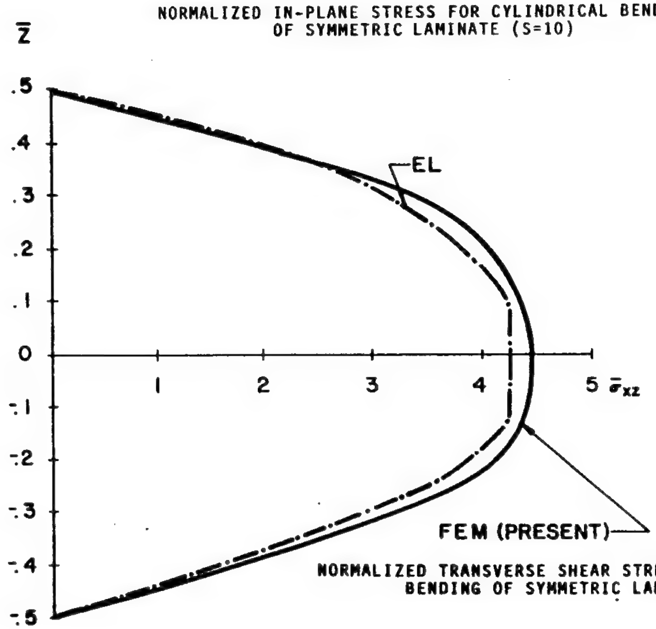
NORMALIZED IN-PLANE STRESS FOR CYLINDRICAL BENDING OF SYMMETRIC LAMINATE (S=4)



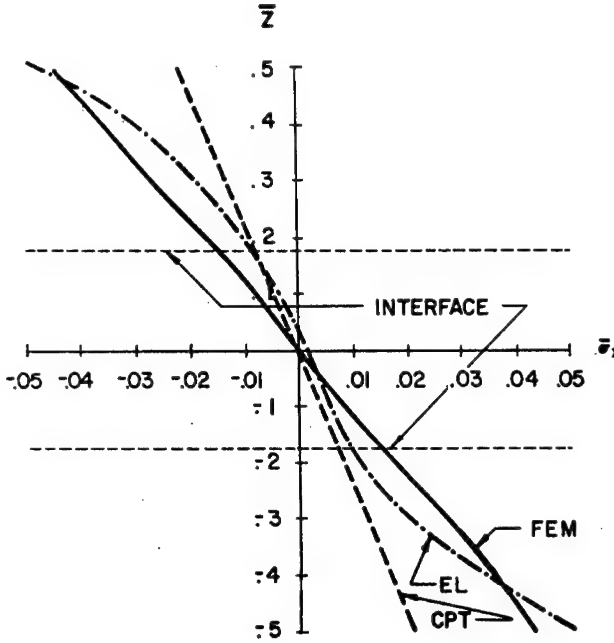
NORMALIZED IN-PLANE STRESS FOR CYLINDRICAL BENDING OF SYMMETRIC LAMINATE (S=10)



NORMALIZED TRANSVERSE SHEAR STRESS FOR CYLINDRICAL BENDING OF SYMMETRIC LAMINATE (S=4)

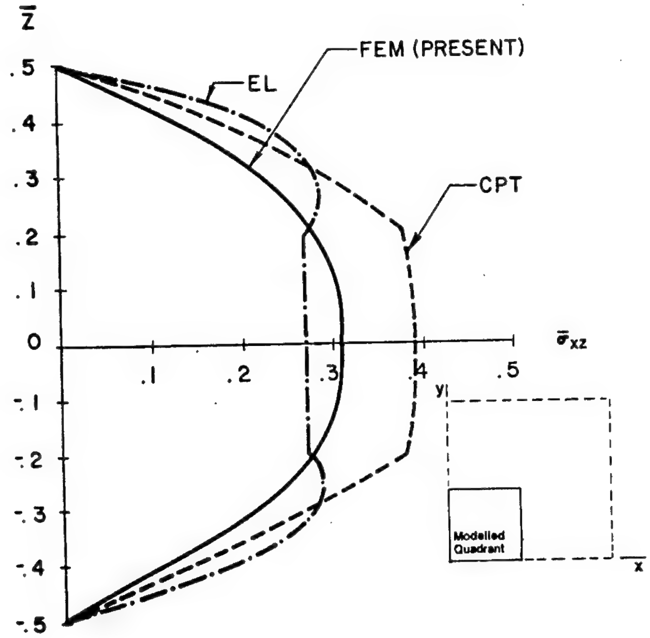


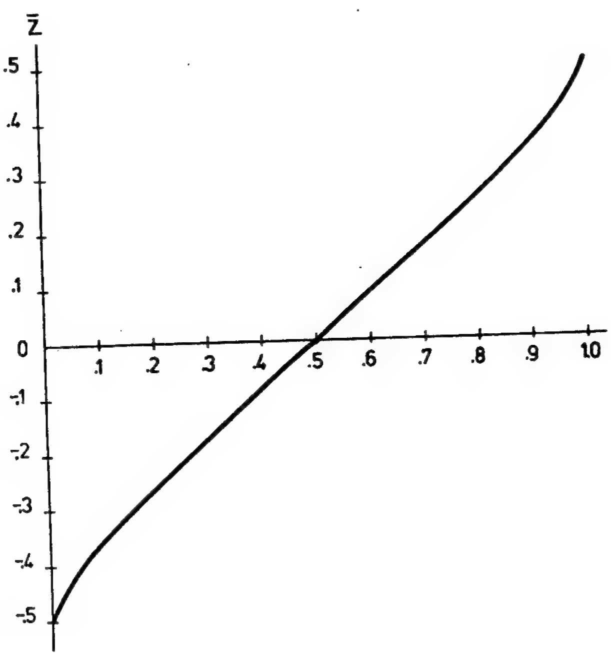
NORMALIZED TRANSVERSE SHEAR STRESS FOR CYLINDRICAL BENDING OF SYMMETRIC LAMINATE (S=10)

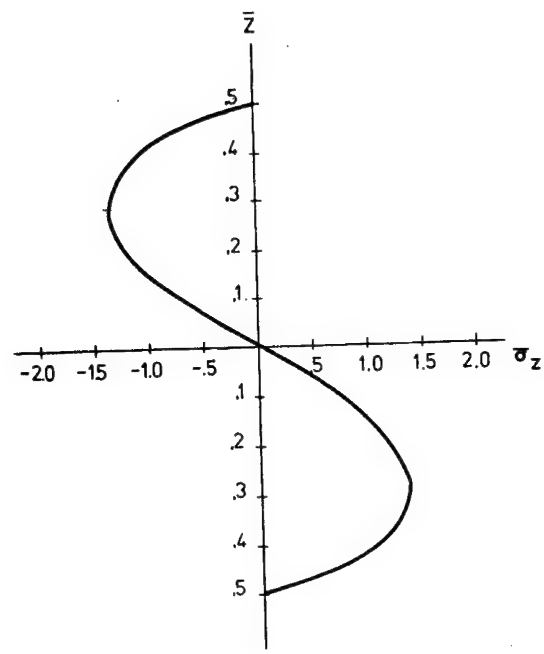


NORMALIZED IN-PLANE SHEAR STRESS FOR SIMPLY SUPPORTED 0-90-0 SQUARE PLATE (x=0, y=0, S=4)

NORMALIZED TRANSVERSE SHEAR STRESS FOR SIMPLY SUPPORTED 0-90-0 SQUARE PLATE (x=0, y=a/2. S=4)







NORMALIZED SHORT-TRANSVERSE NORMAL STRESS FOR SIMPLY SUPPORTED 0-90-0 SQUARE PLATE (x=a/2, y=a/2)

NORMALIZED SHORT-TRANSVERSE NORMAL STRESS FOR SIMPLY SUPPORTED 0-90-0 SQUARE PLATE x=a/2, y=0)

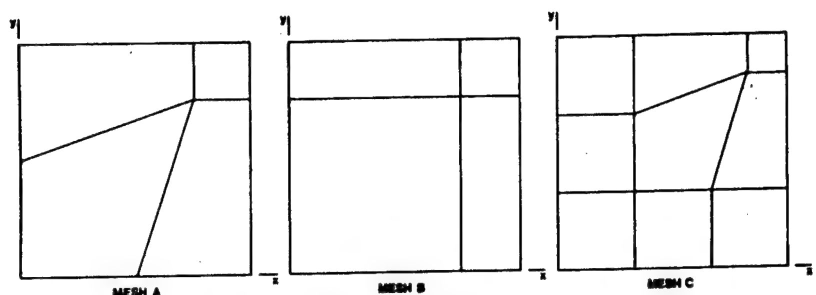
NORMALIZED STRESSES AND DISPLACEMENTS

FOR SIMPLY SUPPORTED SQUARE 0-90-0 PLATE

WITH SINUSOIDAL LOAD

S	Approach	$\bar{\sigma}_x$ $(\frac{a}{2}, \frac{a}{2}, \pm\frac{h}{2})$	$\bar{\sigma}_y$ $(\frac{a}{2}, \frac{a}{2}, \pm\frac{h}{6})$	$\bar{\sigma}_{xy}$ $(0, 0, \pm\frac{h}{2})$	$\bar{\sigma}_{xz}$ $(0, \frac{a}{2}, 0)$	$\bar{\sigma}_{yz}$ $(\frac{a}{2}, 0, 0)$
4	FEM (2x2 Mesh)	.414	.598	.0465	.296	.238
	FEM (6x6 Mesh)	.391	.572	.0448	.308	.251
	Elasticity	.755	.556	.0505	.282	.217
10	FEM (2x2 Mesh)	.529	.292	.0289	.352	.123
	FEM (6x6 Mesh)	.500	.279	.0280	.369	.130
	Elasticity	.590	.288	.0289	.357	.123
20	FEM (2x2 Mesh)	.560	.206	.0240	.367	.0910
	FEM (6x6 Mesh)	.531	.189	.0233	.387	.0954
	Elasticity	.552	.210	.0234	.385	.0938
50	FEM (2x2 Mesh)	.567	.178	.0226	.374	.0812
	FEM (6x6 Mesh)	.541	.164	.0217	.392	.0843
	Elasticity	.541	.185	.0216	.393	.0842
100	FEM (2x2 Mesh)	.566	.174	.0224	.375	.0803
	FEM (6x6 Mesh)	.542	.167	.0215	.393	.0827
	Elasticity	.539	.181	.0213	.395	.0828
	CPT	.539	.180	.0213	.395	.0823

(DISTORTED MESHES)



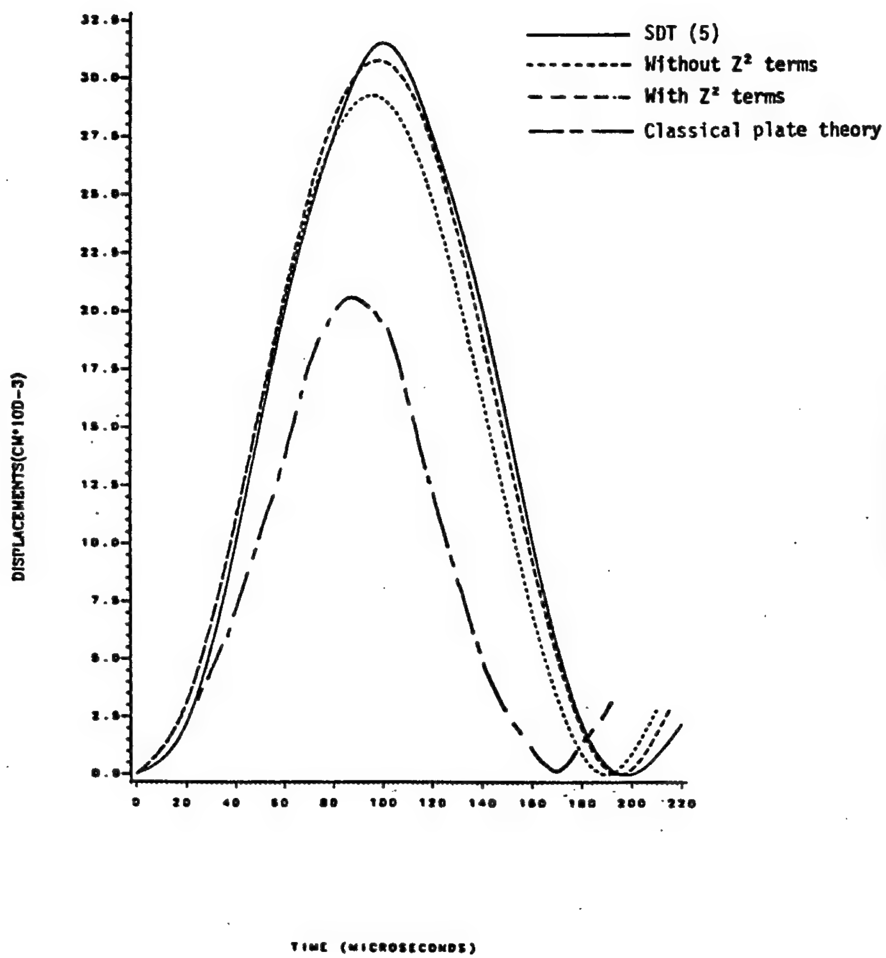
S	Approach	$\bar{\sigma}_x$ $(\frac{a}{2}, \frac{a}{2}, \pm\frac{h}{2})$	$\bar{\sigma}_y$ $(\frac{a}{2}, \frac{a}{2}, \pm\frac{h}{4})$	$\bar{\sigma}_{xy}$ $(0, 0, \pm\frac{h}{2})$	$\bar{\sigma}_{xz}$ $(0, \frac{a}{2}, 0)$	$\bar{\sigma}_{yz}$ $(\frac{a}{2}, 0, 0,)$	$\bar{\omega}$ $(\frac{a}{2}, \frac{a}{2}, 0)$
4	Mesh A	.409	.620	.0525	.128	.122	2.709
	Mesh B	.393	.573	.0485	.254	.206	2.473
	Mesh C	.402	.596	.0472	.290	.229	2.587
	Elasticity	.755	.556	.0505	.282	.217	--
10	Mesh A	.539	.292	.0344	.141	.0686	.866
	Mesh B	.502	.279	.0300	.293	.107	.812
	Mesh C	.530	.286	.0290	.357	.114	.839
	Elasticity	.590	.288	.0289	.357	.123	--
20	Mesh A	.574	.198	.0297	.153	.0552	.545
	Mesh B	.531	.199	.0248	.303	.0792	.529
	Mesh C	.571	.198	.0239	.378	.0880	.539
	Elasticity	.552	.210	.0234	.385	.0938	--
50	Mesh A	.585	.174	.0300	.180	.0516	.443
	Mesh B	.533	.178	.0226	.304	.0701	.445
	Mesh C	.587	.177	.0221	.378	.0955	.449
	Elasticity	.541	.185	.0216	.393	.0842	--
100	Mesh A	.591	.190	.0312	.200	.0363	.418
	Mesh B	.531	.176	.0215	.302	.0689	.432
	Mesh C	.596	.180	.0212	.359	.112	.433
	Elasticity	.539	.181	.0213	.395	.0828	--
	CPT	.539	.180	.0213	.395	.0823	

NORMALIZED STRESSES AND DISPLACEMENTS FOR SIMPLY SUPPORTED SQUARE 0-90-90-0 PLATE WITH SINUSOIDAL LOAD

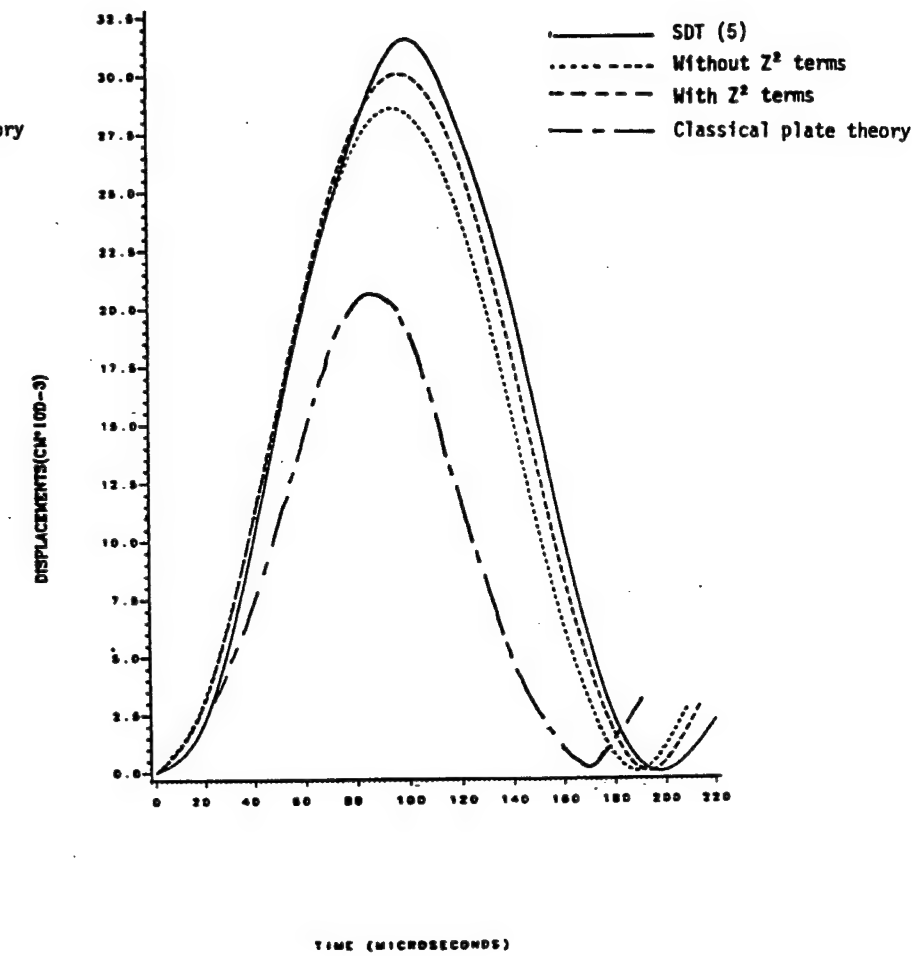
S	Approach	$\bar{\sigma}_x$ $(\frac{a}{2}, \frac{a}{2}, \pm\frac{h}{2})$	$\bar{\sigma}_y$ $(\frac{a}{2}, \frac{a}{2}, \pm\frac{h}{4})$	$\bar{\sigma}_{xy}$ $(0, 0, \pm\frac{h}{2})$	$\bar{\sigma}_{xz}$ $(0, \frac{a}{2}, 0)$	$\bar{\sigma}_{yz}$ $(\frac{a}{2}, 0, 0)$	$\bar{\omega}$ $(\frac{a}{2}, \frac{a}{2}, 0)$
4	FEM (2x2 Mesh)	.409	.646	.0336	.245	.286	5.166
	FEM (6x6 Mesh)	.387	.618	.0326	.256	.302	5.195
	FEM (Reddy)	--	--	--	--	--	--
	Elasticity	.720	.666	.0467	.270	.292	4.491
10	FEM (2x2 Mesh)	.516	.406	.0261	.294	.185	1.762
	FEM (6x6 Mesh)	.488	.388	.0253	.309	.195	1.771
	FEM (Reddy)	.484	.350	.0234	--	--	1.534
	Elasticity	.559	.403	.0276	.301	.196	1.709
20	FEM (2x2 Mesh)	.555	.316	.0233	.313	.147	1.197
	FEM (6x6 Mesh)	.526	.302	.0226	.329	.155	1.203
	FEM (Reddy)	.511	.287	.0214	--	--	1.136
	Elasticity	.543	.309	.0230	.328	.156	1.189
50	FEM (2x2 Mesh)	.566	.283	.0224	.320	.135	1.029
	FEM (6x6 Mesh)	.540	.271	.0216	.336	.140	1.034
	FEM (Reddy)	.520	.265	.0207	--	--	1.019
	Elasticity	.539	.276	.0216	.337	.141	1.031
100	FEM (2x2 Mesh)	.566	.278	.0223	.322	.133	1.003
	FEM (6x6 Mesh)	.542	.266	.0215	.337	.138	1.010
	FEM (Reddy)	.523	.263	.0207	--	--	1.005
	Elasticity	.539	.271	.0214	.339	.139	1.008
	CPT	.539	.269	.0213	.339	.138	1.

NORMALIZED STRESSES AND DISPLACEMENTS FOR SIMPLY SUPPORTED RECTANGULAR 0-90-0 PLATE WITH SINUSOIDAL LOAD

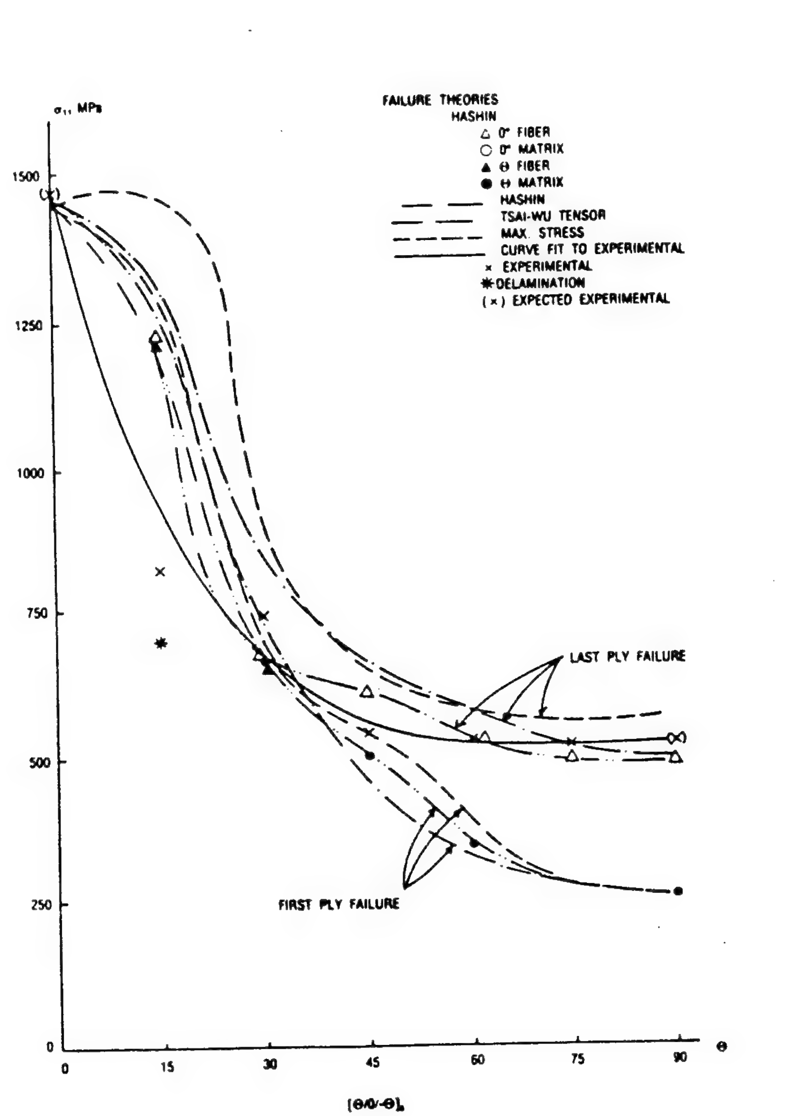
S	Approach	$\bar{\sigma}_x$ $(\frac{a}{2}, \frac{b}{2}, \pm\frac{h}{2})$	$\bar{\sigma}_y$ $(\frac{a}{2}, \frac{b}{2}, \pm\frac{h}{6})$	$\bar{\sigma}_{xy}$ $(0, 0, \pm\frac{h}{2})$	$\bar{\sigma}_{xz}$ $(0, \frac{b}{2}, 0)$	$\bar{\sigma}_{yz}$ $(\frac{a}{2}, 0, 0)$	$\bar{\omega}$ $(\frac{a}{2}, \frac{b}{2}\ 0)$
4	FEM (2x2 Mesh)	.657	.130	.0296	.410	.0357	3.55
	FEM (6x6 Mesh)	.612	.126	.0284	.431	.0391	3.58
	FEM (Reddy)	--	--	--	--	--	--
	Elasticity	.726	.119	.0281	420	.0334	2.82
10	FEM (2x2 Mesh)	.669	.0431	.0125	.414	.0143	.995
	FEM (6x6 Mesh)	.625	.0421	.0121	.436	.0160	1.00
	FEM (Reddy)	.603	.0364	.0102	--	--	.802
	Elasticity	.725	.0435	.0123	.420	.0152	.919
20	FEM (2x2 Mesh	.666	.0286	.00959	.415	.0112	.625
	FEM (6x6 Mesh)	.628	.0278	.00928	.437	.0121	.629
	FEM (Reddy)	.605	.0276	.0086	--	--	.578
	Elasticity	.650	.0299	.0093	.434	.0119	.610
50	FEM (2x2 Mesh)	.659	.0252	.00878	.416	.0113	.521
	FEM (6x6 Mesh)	.629	.0237	.00848	.437	.0110	.524
	FEM (Reddy)	.604	.0251	.0081	--	--	.515
	Elasticity	.628	.0259	.0084	.439	.0110	.520
100	FEM (2x2 Mesh)	.657	.0259	.00856	.416	.0127	.507
	FEM (6x6 Mesh)	.628	.0231	.00837	.437	.0108	.509
	FEM (Reddy)	.603	.0253	.0080	--	--	.506
	Elasticity	.624	.0253	.0083	.439	.0108	.508
	CPT	.623	.0252	.0083	.440	.0108	.503



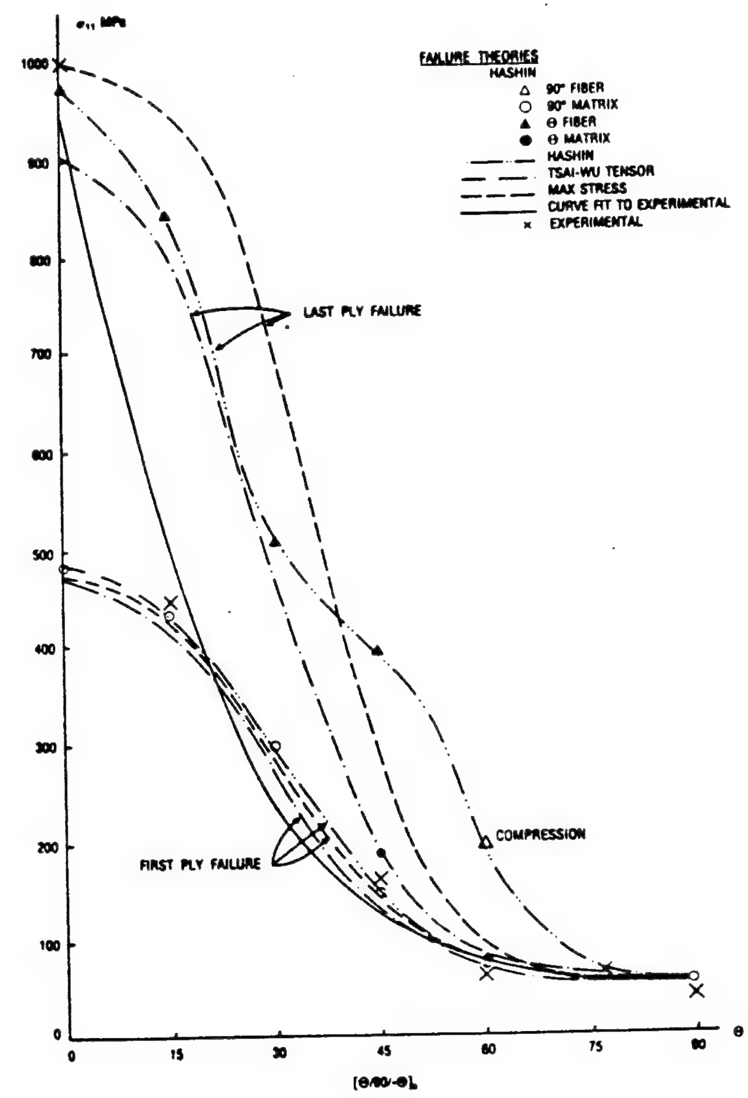
ONE-MODE DISPLACEMENT VS. TIME RESPONSE OF A FINE-MESH (0/90) LAYUP SQUARE PLATE UNDER SUDDENLY-APPLIED SINUSOIDAL LOADING



ONE-MODE DISPLACEMENT VS. TIME RESPONSE OF A COARSE-MESH (0/90) LAYUP SQUARE PLATE UNDER SUDDENLY-APPLIED SINUSOIDAL LOADING



FAILURE CURVES FOR [θ/0/-θ] TRI-DIRECTIONAL LAMINATES DUE TO IN-PLANE LOADING



FAILURE CURVES FOR [θ/90/-θ] TRI-DIRECTIONAL LAMINATES DUE TO IN-PLANE LOADING

RESIDUAL STRENGTH OF THICK FILAMENT-WOUND ROCKET MOTOR CASE AFTER LOW-VELOCITY IMPACTS

C. C. Poe and W. Illg

NASA Langley Research Center
Hampton, Virginia, 23665

and

D. P. Garber

Kentron Technical Center
Hampton, Virginia 23666

ABSTRACT

NASA is making filament-wound rocket motor cases (FWC) for the space shuttle to replace the existing steel cases for certain missions that require a smaller structural mass. The cases are wound with AS-4 graphite fiber impregnated with an epoxy resin and are about 1.4 inches or more thick. They are 12 feet in diameter and about 25 feet long. Each motor is made of four of these cases joined by steel rings, which are mechanically attached by steel pins. Composite laminates, unlike metals, can be damaged relatively easily by low-velocity impacts from objects like dropped tools. Thus, NASA is addressing low-velocity impact in the fracture control plan of the FWC.

Most available impact data come from thin laminate tests. These test results indicate that the damage can cause significant strength loss, even with barely visible evidence of damage. However, the strength loss may decrease with the square root of thickness (ref. 1). Consequently, the strength of the thick FWC may not be seriously reduced by low-velocity impacts. Because there is no data base on very thick laminates to verify this trend, a test program is being conducted by Langley Research Center to determine the residual strength of the thick FWC after low-velocity impacts. These results will supplement results of tests by the contractor on quarter-scale and full-scale cases.

The investigation is planned to give results quickly for inclusion in a fracture control plan for the FWC. The impact tests will be conducted on a representative filament-wound laminate. The laminate will be supported to simulate a fueled (stiff) and an empty (flexible) condition. Impactors of various kinetic energy, mass, and shape will be used. The conditions that give minimum visual evidence of damage will be emphasized. The capability to characterize impact damage with various nondestructive evaluation (NDE) methods will be evaluated. After the impacts, coupons will be cut from the laminate and loaded axially in tension to determine the residual strength. Coupons with surface cracks will also be tested to determine if impact damage, as characterized by NDE, can be equated to surface cracks of the same depth. In this regard, the capability of conventional fracture mechanics to predict the strength with surface cracks will be evaluated. This talk describes the test program and the results obtained to date.

REFERENCES

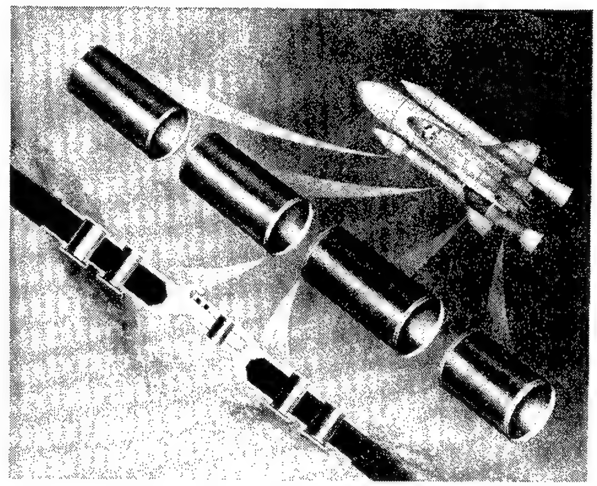
1. Husman, G. E.; Whitney, J. M.; and Halpin, J. C.: Residual Strength Characterization of Laminated Composites Subjected to Impact Loading. Technical Report AFML-TR-73-309, February 1974.

RESIDUAL STRENGTH OF THICK FILAMENT-WOUND ROCKET MOTOR CASE AFTER LOW-VELOCITY IMPACTS

C.C. Poe, Jr., and W. Illg
Langley Research Center

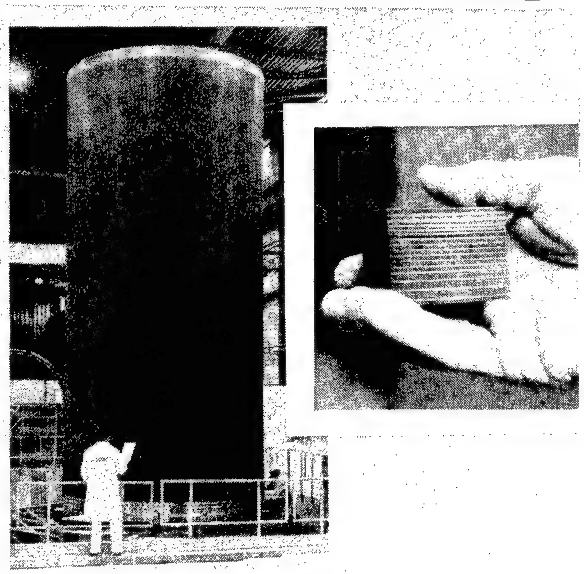
D.P. Garber
Kentron Technical Center

SPACE SHUTTLE SOLID ROCKET MOTOR



HERCULES

FILAMENT-WOUND CASE

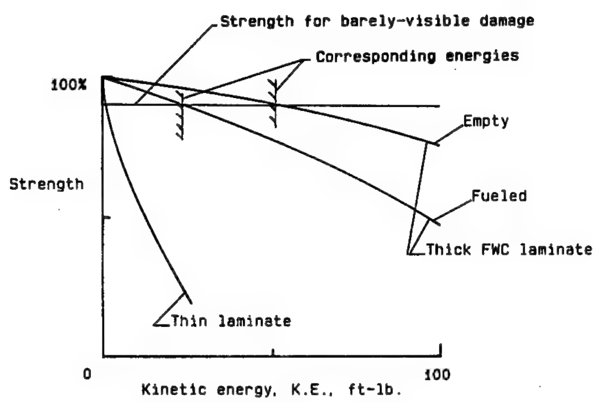


OBJECTIVES

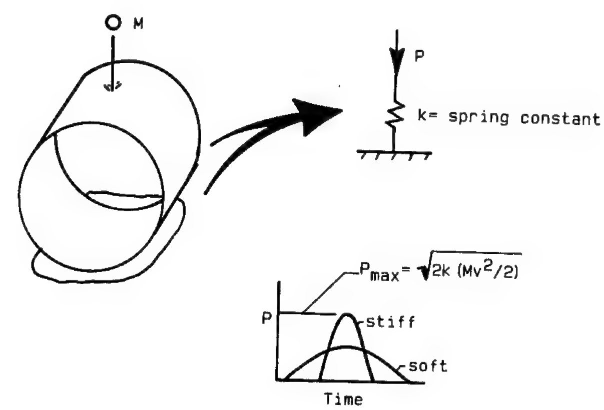
- *Determine strength-loss for damage due to low-velocity impacts, especially for barely-visible damage.*
- *Relate strength-loss to damage-size.*
- *Identify NDE methods that best reveal damage-size.*
- *Relate strength for surface-cracks to strength after impacts.*

Note: These objectives must be achieved quickly to get into a fracture-control plan.

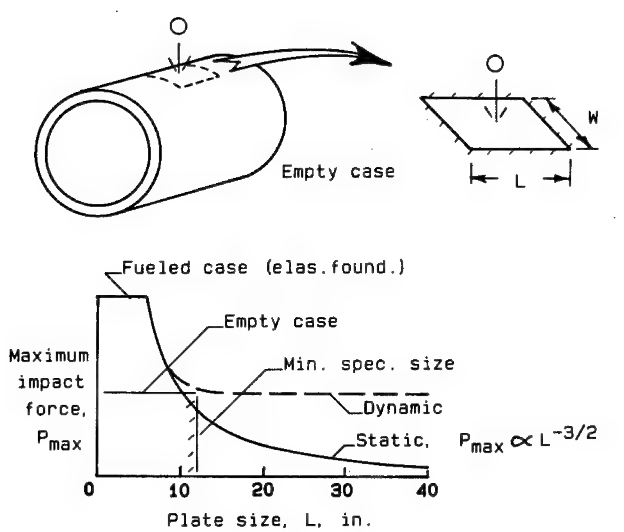
STRENGTH OF IMPACTED FWC



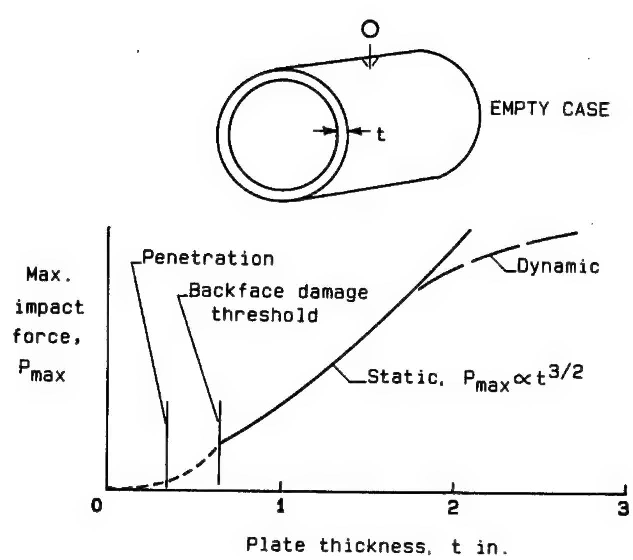
QUASI-STATIC ANALYSIS



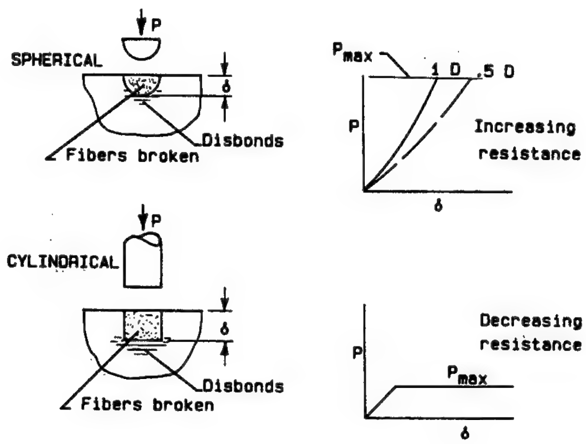
HOW LARGE SHOULD THE SPECIMEN BE ?



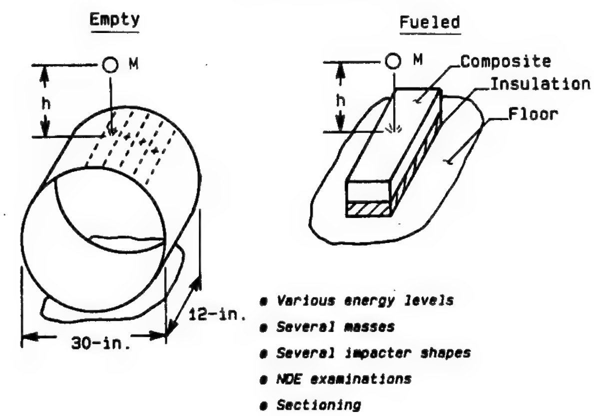
CAN THICKNESS BE SCALED ?



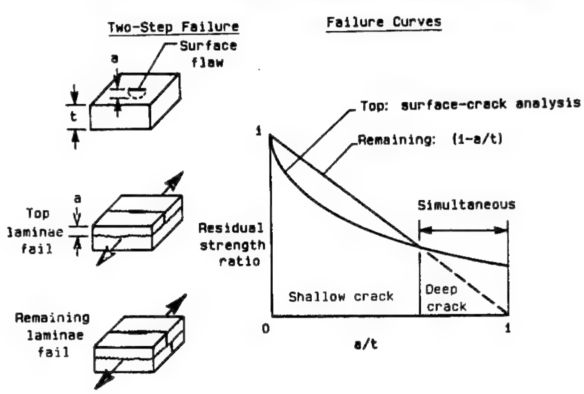
IMPACTOR SHAPE AFFECTS DAMAGE



IMPACT TESTS



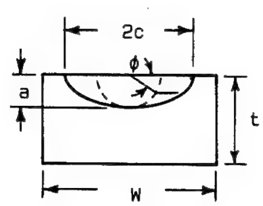
TWO-STEP FRACTURE FOR PART-THROUGH CRACKS



STRENGTH PREDICTIONS

Note: K varies along crack front.

$$S_c = \frac{K_Q (\pi a/Q)^{-1/2}}{F(a/t, a/c, c/w, \phi)}$$



where: $Q = 1 + 1.464\,(a/c)^{1.65}$ for $a/c \leq 1$

$= 1 + 1.464\,(c/a)^{1.65}$ for $a/c > 1$

and F is given by Newman and Raju.

FRACTURE TOUGHNESS PREDICTIONS

$$K_Q = \left(\frac{Q_c}{\epsilon_{tuf}}\right) E_x \epsilon_{tuf} / (1 - \nu_{xy}\sqrt{E_y/E_x})$$

$Q_c/\epsilon_{tuf} = 0.30\ \sqrt{in.}$ (previous work)

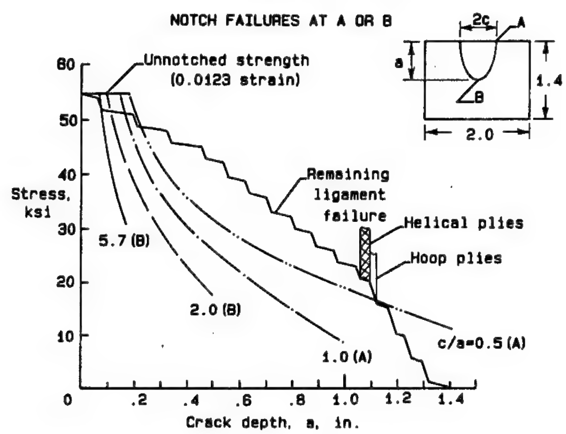
$E_x \epsilon_{tuf} = F_{tu}$ (assumption)

$F_{tu} = 54.8$ ksi ($\epsilon_{tuf} = 0.0123$) (Tensile test)

$E_x = 4.44\times10^6$ psi
$E_y = 5.66\times10^6$ psi
$\nu_{xy} = 0.351$
} (lamination theory- no bending)

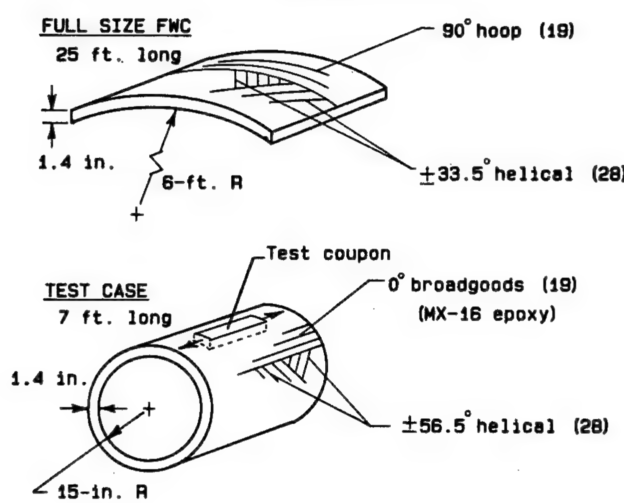
$\therefore\ K_Q = 27.2\ \text{ksi}\ \sqrt{in.}$

PREDICTED FWC TENSILE STRENGTHS
FOR SURFACE CRACKS



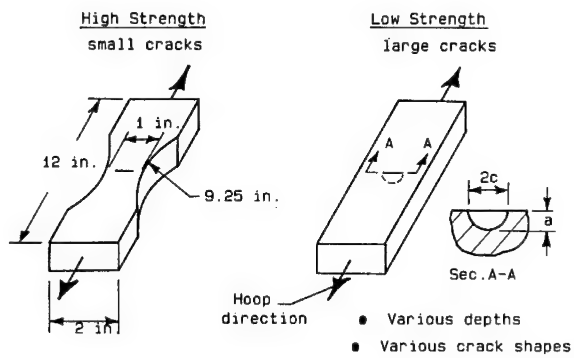
TESTS

MATERIALS FOR TEST
AS4W/BRF55A



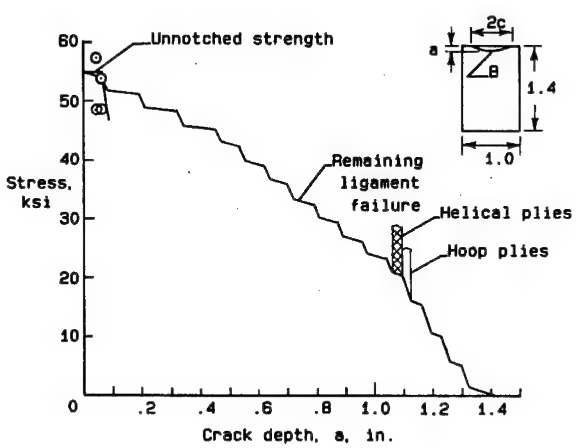
FRACTURE TEST RESULTS

SHAPES OF FRACTURE SPECIMENS



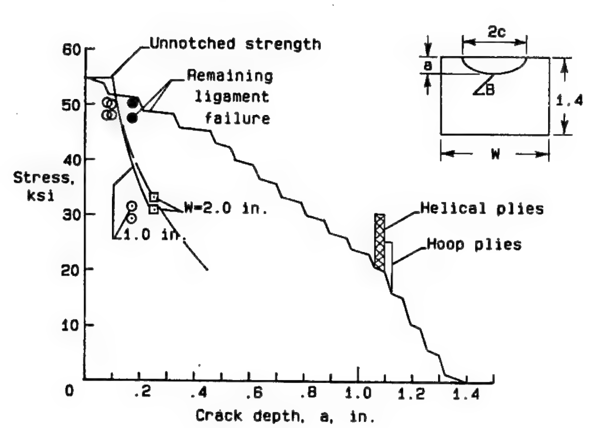
TENSILE TEST RESULTS FOR SURFACE CRACKS

NOTCH FAILURES AT B, c/a=5.7

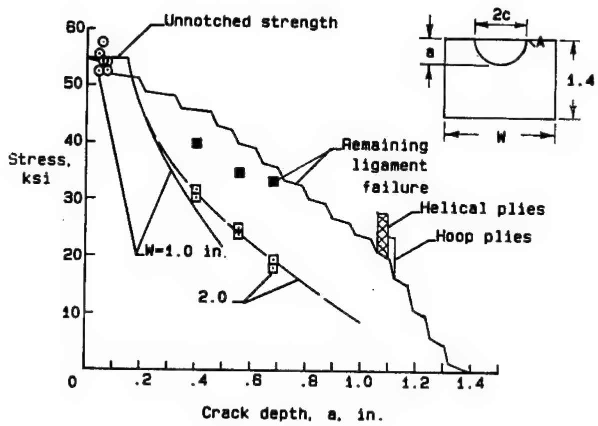


TENSILE TEST RESULTS FOR SURFACE CRACKS

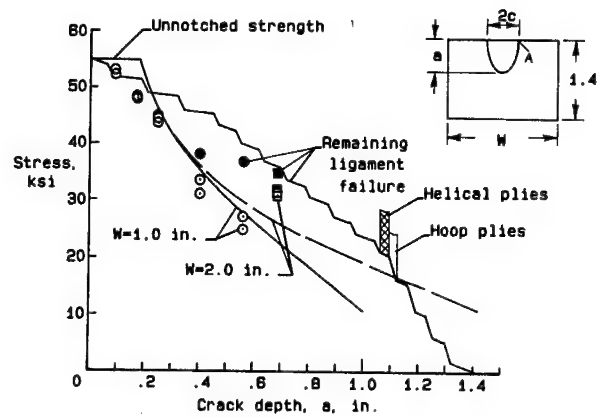
NOTCH FAILURES AT B, c/a=2.0



TENSILE TEST RESULTS FOR SURFACE CRACKS
Notch-failures at A, c/a=1.0

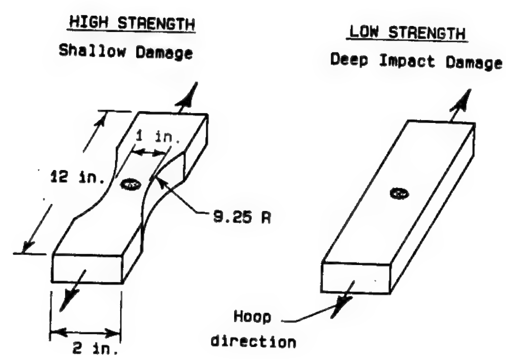


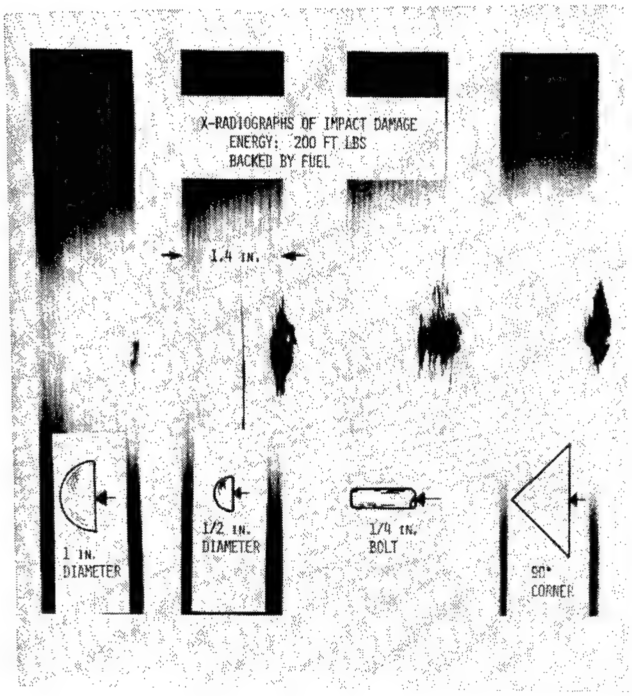
TENSILE TEST RESULTS FOR SURFACE CRACKS
NOTCH FAILURES AT A, c/a=0.5



IMPACT TEST RESULTS

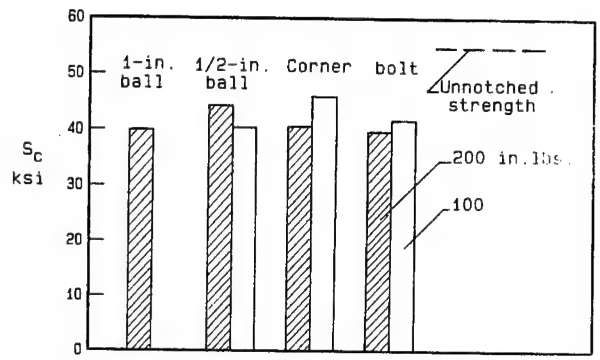
SHAPES OF RESIDUAL STRENGTH IMPACT SPECIMENS





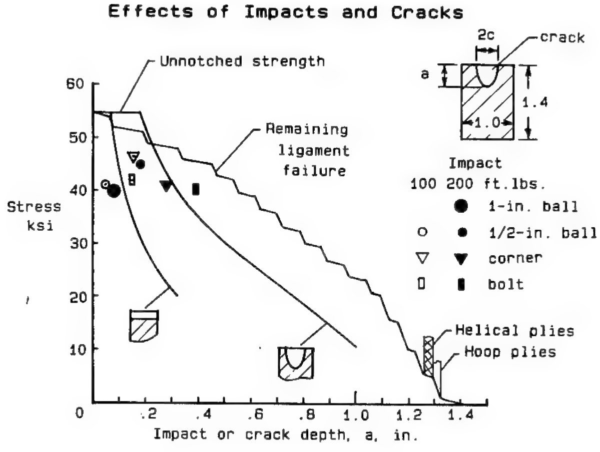
RESIDUAL TENSILE STRENGTHS

Effects of Shapes and Energies



RESIDUAL TENSILE STRENGTHS

Effects of Impacts and Cracks



TEST DIFFICULTIES

- *Large failure-loads require high squeeze-loads to avoid slippage.*
- *Premature failures occur at grips due to high squeeze stresses.*
- *Axial strains vary across test-section due to:*
  Layup asymmetry,
  Grip misalignment.
- *Load path becomes eccentric when first ligament fails.*
- *Specimen width is limited by testing machine's load-capacity.*
- *Narrowness of specimen causes edge-effects during impacts and tensile tests.*
- *Cylindrical curvature complicates gripping.*

BOLTED JOINTS IN COMPOSITE STRUCTURES: DESIGN, ANALYSIS AND VERIFICATION

R. L. Ramkumar, E. S. Saether and E. W. Tossavainen

Northrop Corporation, Aircraft Division
Hawthorne, California 90250

V. B. Venkayya and M. Sobota
Air Force Wright Aeronautical Laboratories
Air Force Base, Ohio 45433

ABSTRACT

Bolted joints are a primary means of load transfer in structures. A clear understanding of the behavior of single fastener joints is a prerequiste for the design of general multifastener joints. Only limited success has previously been achieved in analytically predicting the strength of single fastener joints, and the design of bolted joints and the selection of fastener patterns have hitherto been based mostly on experimental results.

The objectives of the current program are to develop improved analytical methods for strength and life prediction of bolted joints, to verify the developed analyses through a series of experiments, and to develop a comprehensive, design-oriented guide for bolted joints in composite structures.

Most of the work done to date on this program has been on single fastener joints. To predict approximately the three-dimensional stress distribution in a single fastener joint, the methodology developed under Northrop Independent Research and Development Programs (References 1 and 2) were modified for current use. In Reference 1, a boundary collocation technique was used to develop a two-dimensional analysis to account for finite geometry effects in an anisotropic plate with a fastener hole. The validity of this analysis was established by comparing predicted stress distributions with finite element solutions. In Reference 2, the bearing load distribution through the thickness was approximated by modeling the fastener as a Timoshenko beam on an elastic foundation. The effects of fastener flexibility, in bending and in shear, were demonstrated by comparing analytical predictions with two-dimensional plate solutions. The two analyses were incorporated into a progressive failure procedure to predict the joint strength.

A total of 450 specimens of various joint configurations have been tested. The specific joint parameters assessed include bolted plate properties and dimensions, laminate layup and stacking sequences, fastener properties and size, fastener torque, joint configuration (single or double lap), load type (static and constant amplitude fatigue), and test environment. Typical results are shown in the following pages. Test results have verified trends established earlier in the literature (Reference 3) and are being correlated with the analytical methodology.

REFERENCES

1. Kudva, N. J. and Madenci, E., "Stress Analysis of Finite Geometry Composite Bolted Joints Using Modified Boundary Colloçation Techniques", Northrop Corporation Report Number NOR 83-198, October 1983.

2. E. Saether and R. L. Ramkumar, "Strength Analysis of Bolted Laminates Accounting for Three Dimensional Effects at the Fastener Location", Northrop Corporation Report Number NOR 84-122, July 1984.

3. Garbo, S. P. and Ogonowski, J. M., "Effect of Variances and Manufacturing Tolerances on the Design Strength and Life of Mechanically Fastened Composite Joints", Volumes 1, 2 and 3, AFWAL-TR-81-3041, April 1981.

BOLTED JOINTS IN COMPOSITE STRUCTURES: DESIGN, ANALYSIS AND VERIFICATION

R.L. RAMKUMAR, E.S. SAETHER AND E.W. TOSSAVAINEN

NORTHROP CORPORATION, AIRCRAFT DIVISION
HAWTHORNE, CALIFORNIA

V.B. VENKAYYA AND M. SOBOTA

AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
WRIGHT-PATTERSON AIR FORCE BASE
DAYTON, OHIO

PROGRAM OBJECTIVES

- DEVELOP ANALYTICAL METHODS FOR STRENGTH AND LIFE PREDICTION OF BOLTED JOINTS IN COMPOSITE STRUCTURES
- VALIDATE ANALYSES THROUGH TESTS ON SINGLE AND MULTI-FASTENER JOINTS, AND ELEMENTS REPRESENTATIVE OF FULL-SCALE STRUCTURES
- DEVELOP A DESIGN GUIDE TO AID IN THE DESIGN OF BOLTED JOINTS IN COMPOSITE STRUCTURES

PROGRAM TASKS

TASK I - DEVELOP ANALYTICAL TECNIQUES FOR SINGLE FASTENER JOINTS

TASK II - DEVELOP ANALYTICAL TECHNIQUES FOR MULTI-FASTENER JOINTS ACCOUNTING FOR STRESS CONCENTRATION INTERACTION DUE TO GEOMETRY CHANGES (CUT-OUTS, WIDTH TAPERING, PLY DROP-OFF, ETC.)

TASK III - CONDUCT FULL SCALE VERIFICATION TESTS

TASK IV - GENERATE A DESIGN GUIDE

TWO-DIMENSIONAL STRESS/STRAIN FIELD IN A BOLTED LAMINATE

COMPATIBILITY EQUATION

$$a_{22}\ F_{,xxxx} - 2a_{26}\ F_{,xxxy} + (2a_{12} + a_{66})F_{,xxyy}$$
$$-2a_{16}\ F_{,xyyy} + a_{11}\ F_{,yyyy} = 0$$

a_{ij} are laminate compliances
F is Airy's stress function

CHARACTERISTIC EQUATION

$$a_{11}\mu^4 - 2a_{16}\mu^3 + (2a_{12} + a_{66})\mu^2 - 2a_{26}\mu + a_{22} = 0$$

$Z_1 = x + \mu_1 y$ and $Z_2 = x + \mu_2 y$ are complex quantities.

Stresses, strains and displacements are expressed in terms of $\phi_1(Z_1)$, $\phi_2(Z_2)$ and their derivatives

ASSUMED SOLUTIONS

$$\phi_1(\xi_1) = \alpha_o\ \ln \xi_1 + \sum_{n=1}^{\infty} (\alpha_{-n}\xi_1^{-n} + \alpha_n' \xi_1^{n})$$

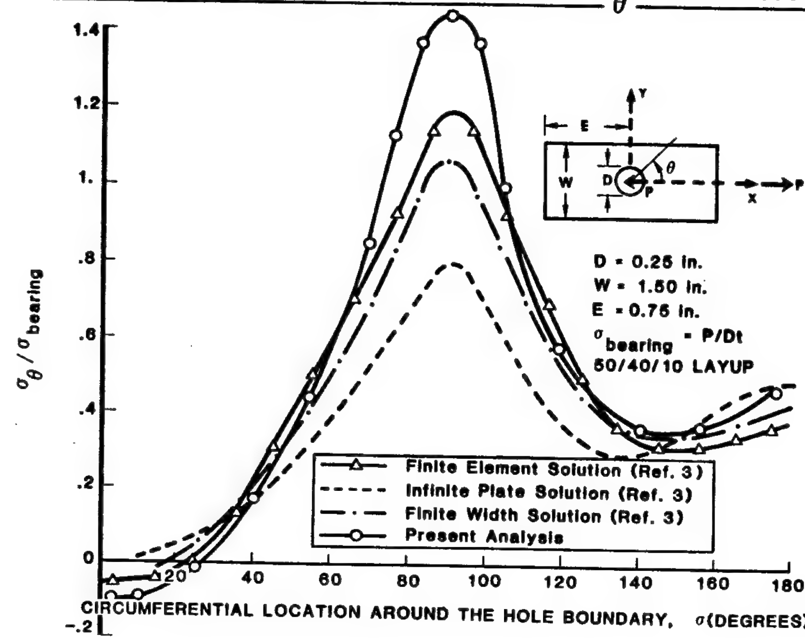
$$\phi_2(\xi_2) = \beta_o\ \ln \xi_2 + \sum_{n=1}^{\infty} (\beta_{-n}\xi_2^{-n} + \beta_n \xi_2^{n})$$

$\xi_1(z_1)$ and $\xi_2(z_2)$ are conformal mapping functions that map the physical region of the bolted plate to the exterior of unit circles.

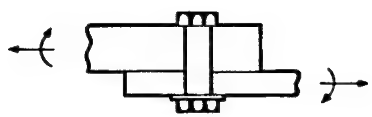
SOLUTION PROCEDURE

- ASSUME A FINITE NUMBER OF TERMS IN THE ϕ_1 AND ϕ_2 SERIES
- IMPOSE SINGLE-VALUED RESTRICTIONS ON DISPLACEMENTS
- SUPPRESS RIGID BODY ROTATION
- SELECT A LARGE NUMBER OF BOUNDARY COLLOCATION POINTS, AND SATISFY BOUNDARY CONDITIONS USING A LEAST SQUARES APPROACH

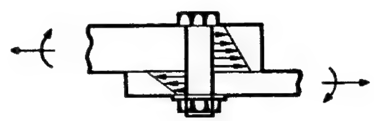
EFFECT OF FINITE GEOMETRY ON σ_θ VARIATION



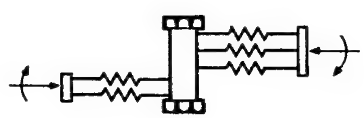
FASTENER MODELING



(a) TYPICAL JOINT IN SINGLE SHEAR



(b) TYPICAL CONTACT FORCE DISTRIBUTION



(c) MATHEMATICAL REPRESENTATION

GOVERNING EQUATION FOR FASTENER ANALYSIS

$$u'''' + \frac{q''}{\lambda GA} - \frac{q}{EI} = 0$$

$q = -ku - \bar{k}\,\bar{u}$

$k = k_1$ for an undamaged ply

$= k_2 = \alpha k_1$ for a partially damaged ply

$= 0$ for a totally damaged ply

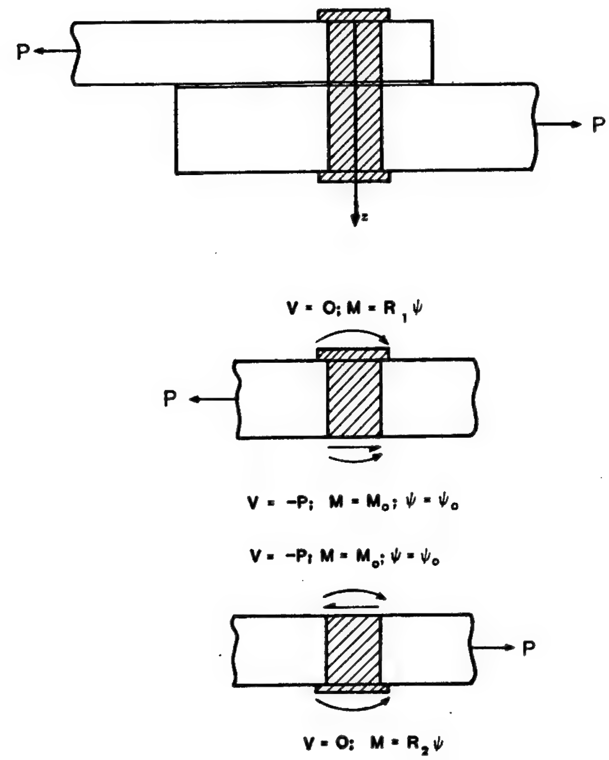
$\bar{k} = k_1 - k_2$ for a partially damaged ply

$= 0$ for a totally damaged ply

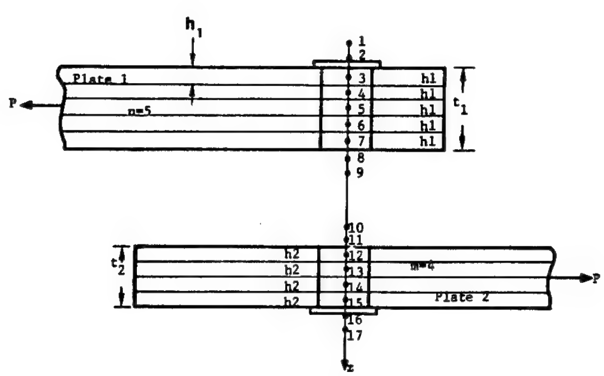
$\bar{u} = 0$ for an undamaged ply

$= u_o$ for a partially damaged ply

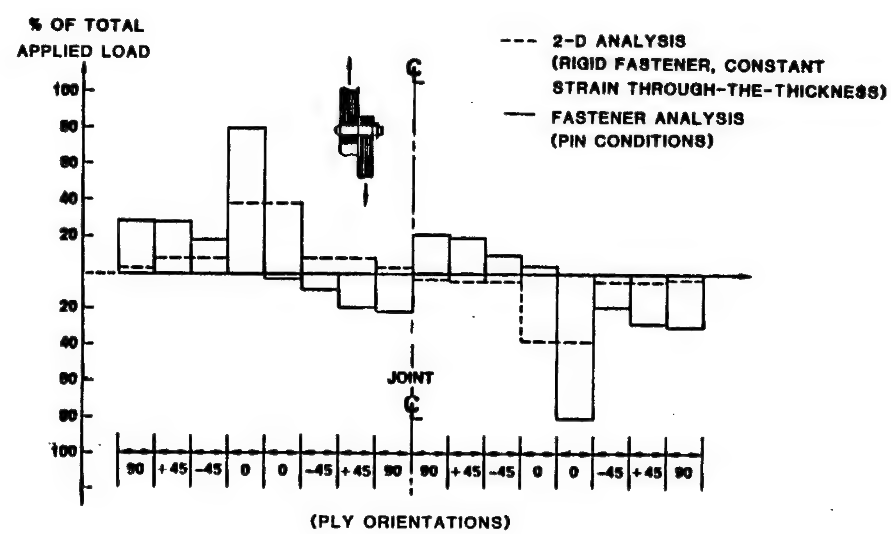
BOUNDARY AND CONTINUITY CONDITIONS FOR A SINGLE SHEAR CONFIGURATION



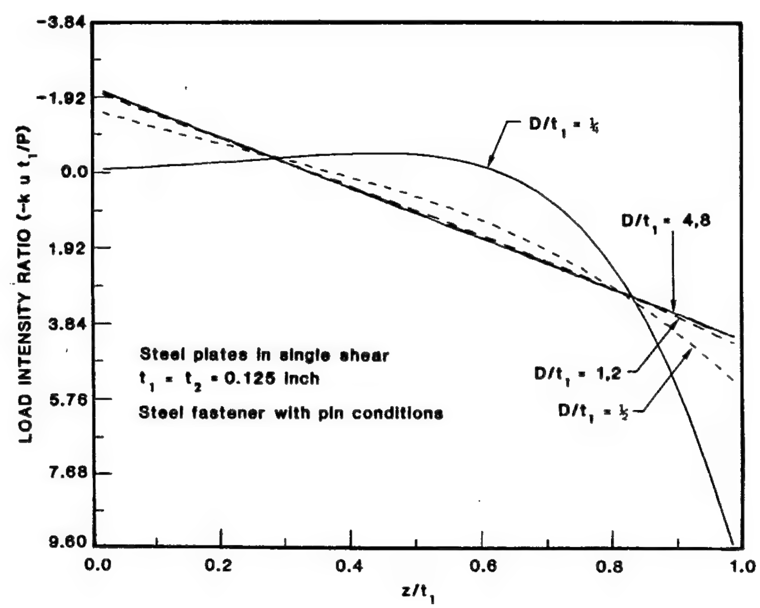
SAMPLE DISCRETIZATION OF THE FASTENER FOR A SINGLE SHEAR CONFIGURATION



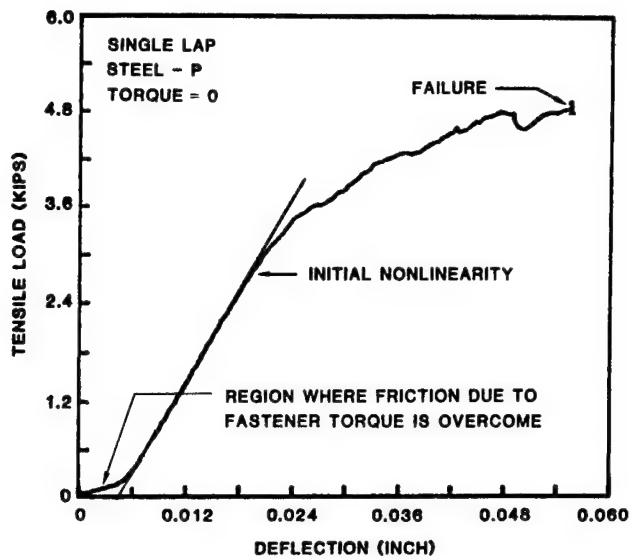
COMPARISON BETWEEN 2-D AND 3-D RESULTS



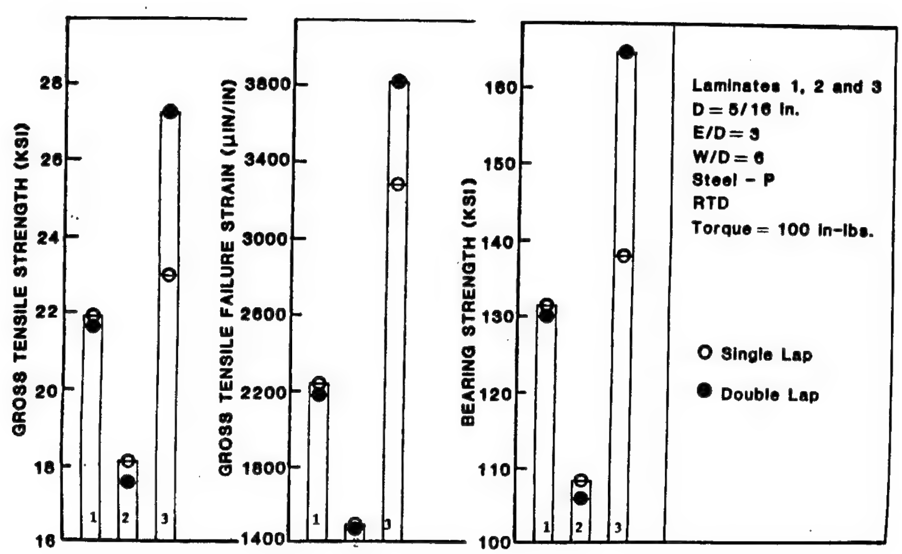
EFFECT OF FASTENER SIZE ON THE LOAD DISTRIBUTION



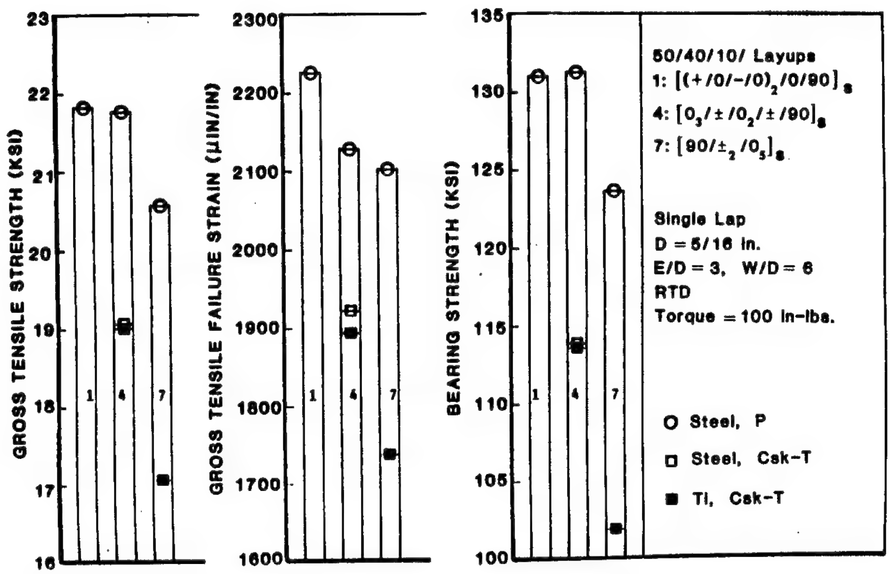
TYPICAL LOAD vs CLIP GAGE DEFLECTION PLOT



DOUBLE LAP vs SINGLE LAP



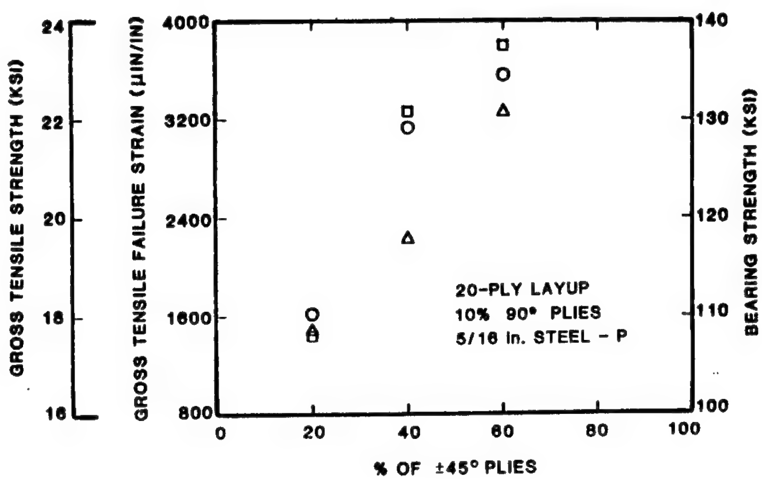
STACKING SEQUENCE EFFECT



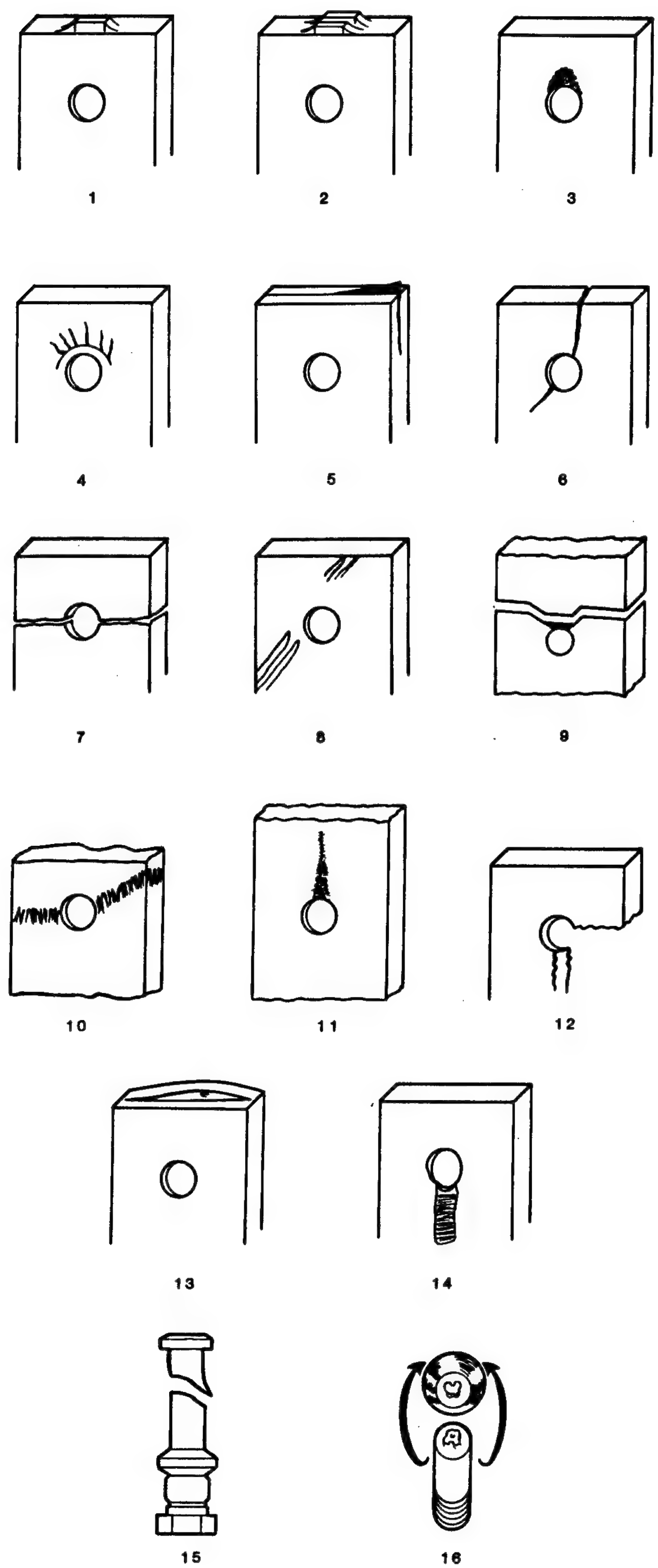
EFFECT OF LAYUP

SINGLE LAP, COMPOSITE-TO-ALUMINUM, E/D = 3, W/D = 6, RTD

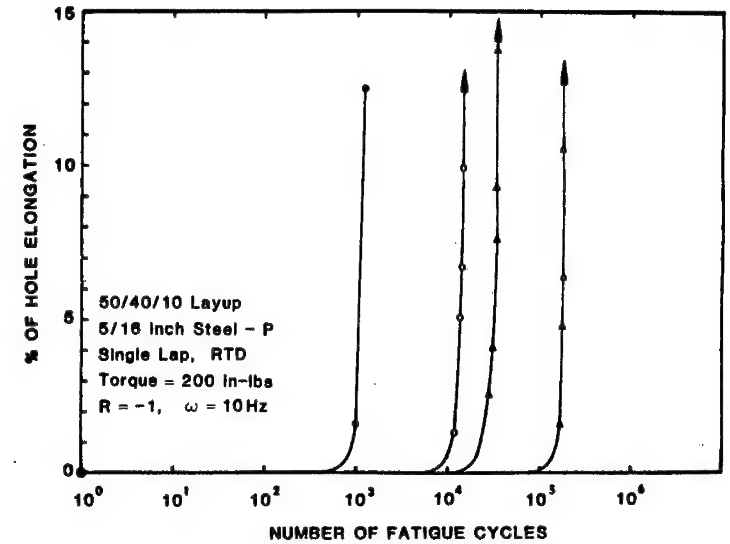
O Gross Tensile Strength △ Gross Tensile Failure Strain □ Bearing Strength



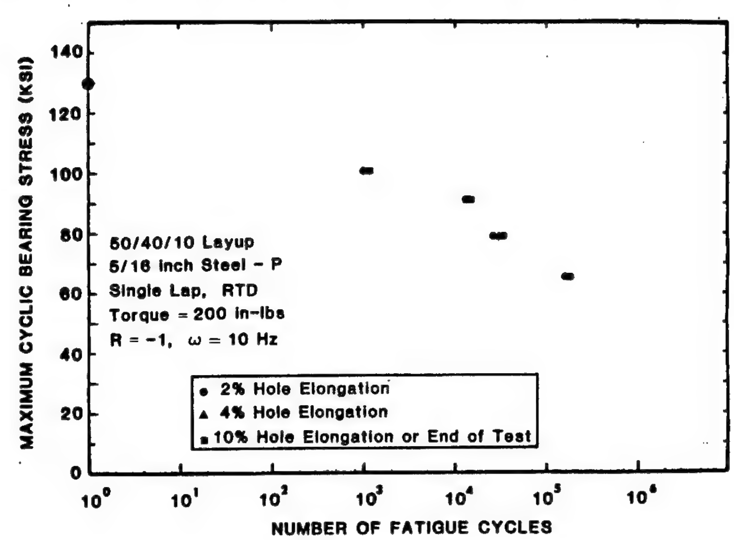
OBSERVED FAILURE MODES



HOLE ELONGATION WITH FATIGUE CYCLES



NUMBER OF CYCLES TO SPECIFIED HOLE ELONGATION vs MAXIMUM CYCLIC BEARING STRESS



REMAINING PROGRAM TASKS

(1) MULTIFASTENER ANALYSIS

(2) TESTS ON MULTIFASTENER COMPOSITE-TO-METAL JOINT SPECIMENS

(3) ANALYSIS OF ELEMENT TEST RESULTS

(4) DEVELOPMENT OF THE DESIGN GUIDE

DESIGN AND ANALYSIS OF POSTBUCKLED COMPOSITE AND METAL STRUCTURES

B. L. Agarwal, R. B. Deo

Northrop Corporation, Aircraft Division
Hawthorne, California 90250

ABSTRACT

Northrop has been actively involved in postbuckling research on government-sponsored programs and on continuing independent research and development (IR&D) programs. Under two Air Force funded contracts, Northrop is developing analysis methods and design methodology for metal and composite structures subjected to shear or compression loading. A design guide is also being developed under the same contract at Northrop. Northrop has a long history of Navy-sponsored contracts in the area of postbuckling. Under Contract N62269-79-C-0461, Northrop examined the postbuckling behavior of multi-stiffened composite shear panels subjected to constant amplitude fatigue loading. The postbuckling behavior of curved stiffened compression panels (under Contract N00019-79-C-0549) and the effect of spectrum fatigue loading on shear panels (under Contract N62269-81-C-0321) were also investigated. Laser irradiation effects on postbuckled panels were examined under Contract N00014-81-C-2281. Northrop has developed analysis methods and test procedures for combined loads under recent IR&D programs and is continuing to develop analysis methods in IR&D programs which supplement the work in current literature.

Extensive information is now available to develop a design methodology for postbuckled composite and metal panels. A concerted effort was undertaken at Northrop to collect all the information, identify and fill any gaps, and make available to the designers a methodology to design postbuckled aircraft structures to achieve lighter weight, reduced cost, and longer fatigue life. This paper highlights the state-of-the-art in the design and analysis of postbuckled structures.

ISSUES

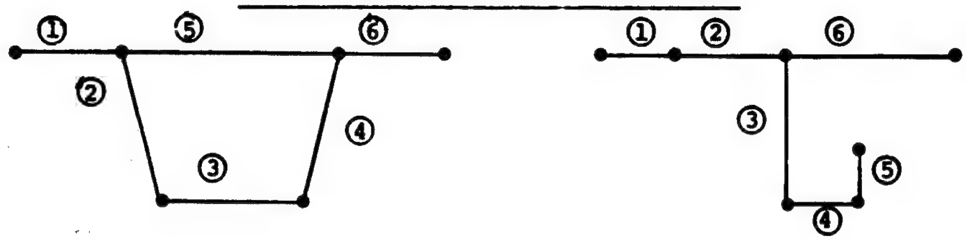
- DESIGN METHODS
- DURABILITY
- DAMAGE TOLERANCE
- ENVIRONMENTAL EFFECTS
- REPAIR TECHNIQUES

COMPOSITE COMPRESSION PANEL DESIGN

- DETERMINE BUCKLING STRAINS FOR
  - EULER BUCKLING
  - SKIN BUCKLING
  - STIFFENER CRIPPLING
- DETERMINE FAILURE LOADS DUE TO
  - EULER BUCKLING
  - STIFFENER CRIPPLING
  - STIFFENER/WEB SEPARATION

$$\varepsilon_{SS} = 0.4498\, \varepsilon_{CR}^{SK} \left(\frac{\varepsilon_{CU}}{\varepsilon_{CR}^{SK}} \right)^{0.72715}$$

FAILURE DUE TO STIFFENER CRIPPLING



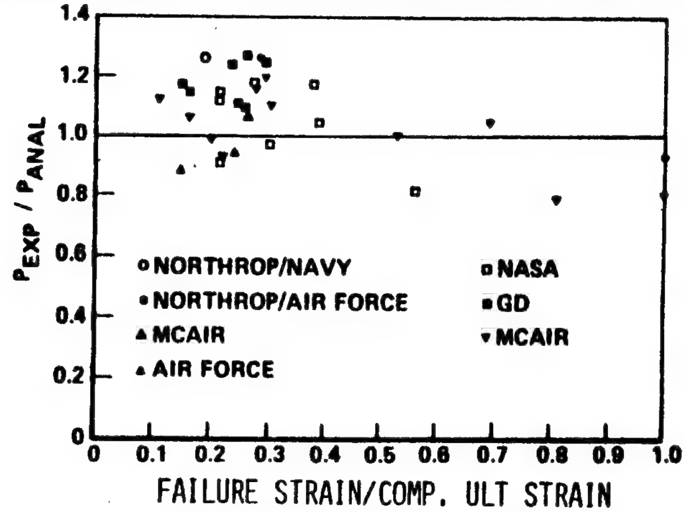
- DETERMINE TWO ELEMENTS WITH THE LOWEST CRIPPLING STRAIN

$$\varepsilon_{CC} = \alpha \;\; \varepsilon_{CR} \left(\frac{\varepsilon_{CU}}{\varepsilon_{CR}} \right)^{\beta}$$

VALUES OF α AND β DEPEND ON SUPPORT CONDITIONS.

- IF THE MEMBER WITH LOWEST CRIPPLING STRAIN IS NORMAL TO THE AXIS OF LEAST BENDING STIFFNESS THE FAILURE LOAD IS BASED UPON THE LOWEST CRIPPLING STRAIN. IF NOT THE PANEL CARRIES ADDITIONAL LOAD AND FAILURE LOAD IS BASED UPON THE SECOND LOWEST CRIPPLING STRAIN.

COMPARISON OF ANALYTICAL AND EXPERIMENTAL DATA FOR COMPOSITE COMPRESSION PANELS



COMPARISON OF ANALYTICAL AND EXPERIMENTAL DATA (COMPOSITE SHEAR PANELS)

- ANALYSIS BASED UPON TENSION FIELD THEORY MODIFIED TO ACCOUNT FOR MATERIAL ANISOTROPY - FORCED CRIPPLING INCLUDED.
- TEST DATA CONSISTS OF AVAILABLE DATA FROM THE FOLLOWING PROGRAMS.
  - NORTHROP/NAVY CONTRACT
  - MCAIR/NAVY CONTRACT
  - LOCKHEED/NAVY CONTRACT
  - GRUMMAN/NORTHROP/NAVY CONTRACT
  - 3 NORTHROP/IRAD PANELS

COMPARISON OF ANALYTICAL AND EXPERIMENTAL DATA (COMPOSITE SHEAR PANELS)

REFERENCE	H_S (INCH)	H_R/H_S	STIFFENER SHAPE	EA_S (MSI)	EI_S LB/IN.^{2} X 10^6	$\frac{\tau_{ULT}}{\tau_{CR}}$	P_{ANL}/P_{EXP}
NORTHROP/NAVY	10	1.5	HAT	2.5	0.40	5	0.93
MCDONNELL/NAVY	6	2.67	HAT	1.8	0.37	9.2	0.91 (1.03*)
LOCKHEED/NAVY	6	3.75	I	1.5	0.90	6	1.08
GRUMMAN/NAVY	7	3.43	HAT	2.9	1.0	6	1.025
NORTHROP/IRAD	13	1.15	HAT	2.9	0.73	10	1.05
NORTHROP/IRAD	13	1.15	I	2.9	0.313	10	0.91
NORTHROP/IRAD	9	1.66	HAT	2.9	0.30	7	0.80

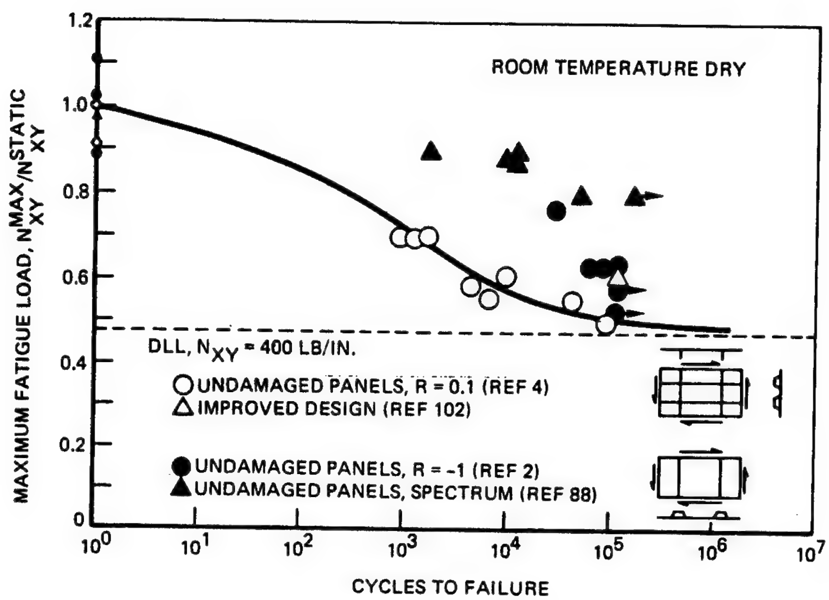
* = FAILURE DUE TO RING CRIPPLING
H_S = STRINGER SPACING
H_R = RING SPACING
EA_S = STRINGER AXIAL STIFFNESS
EI_S = STRINGER BENDING STIFFNESS
τ_{ULT} = ULTIMATE FAILURE STRESS
τ_{CR} = BUCKLING STRESS
P_{ANL} = ANALYTICAL FAILURE LOAD
P_{EXP} = EXPERIMENTAL FAILURE LOAD

STATUS OF FAILURE LOAD ANALYSIS

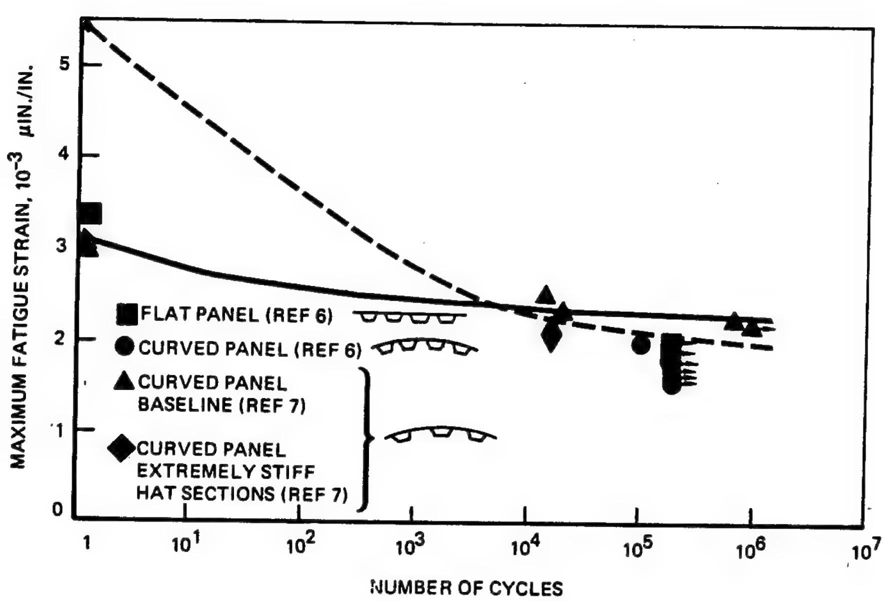
LOADING	METAL PANELS		COMPOSITE PANELS	
	FLAT PANELS	CURVED PANELS	FLAT PANELS	CURVED PANELS
COMPRESSION	√	√	√√	√√
SHEAR	√	√	√√	√√√
COMBINED COMPRESSION SHEAR	√	√	X	X

√ - ALREADY EXISTS
√√ - DEVELOPED UNDER CURRENT AIR FORCE CONTRACT
√√√ - WILL BE DEVELOPED UNDER CURRENT AIR FORCE CONTRACT
X - NEEDS TO BE DEVELOPED

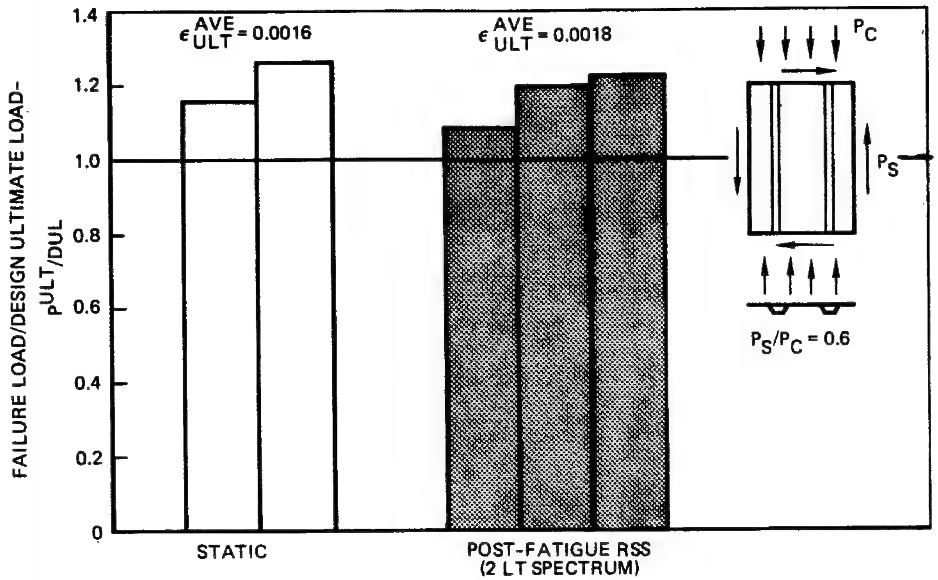
DURABILITY OF POSTBUCKLED PANELS



COMPOSITE SHEAR PANEL FATIGUE RESPONSE

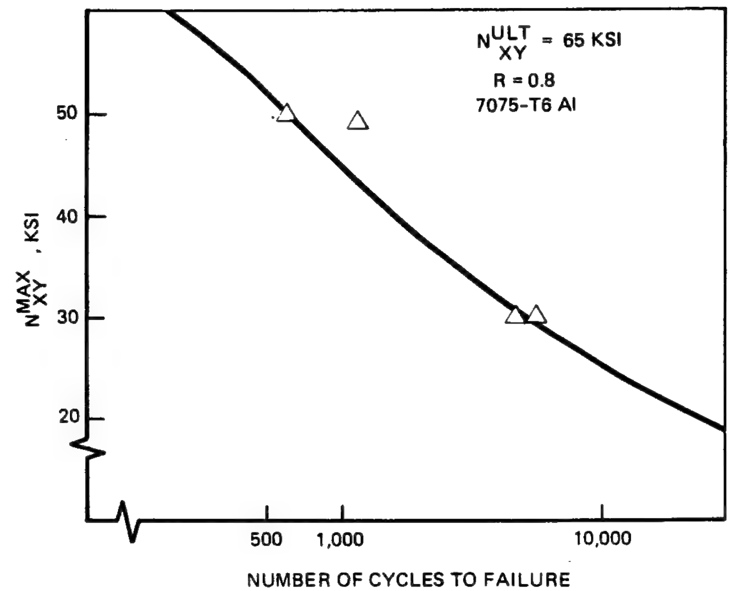


COMPOSITE COMPRESSION PANEL FATIGUE RESPONSE

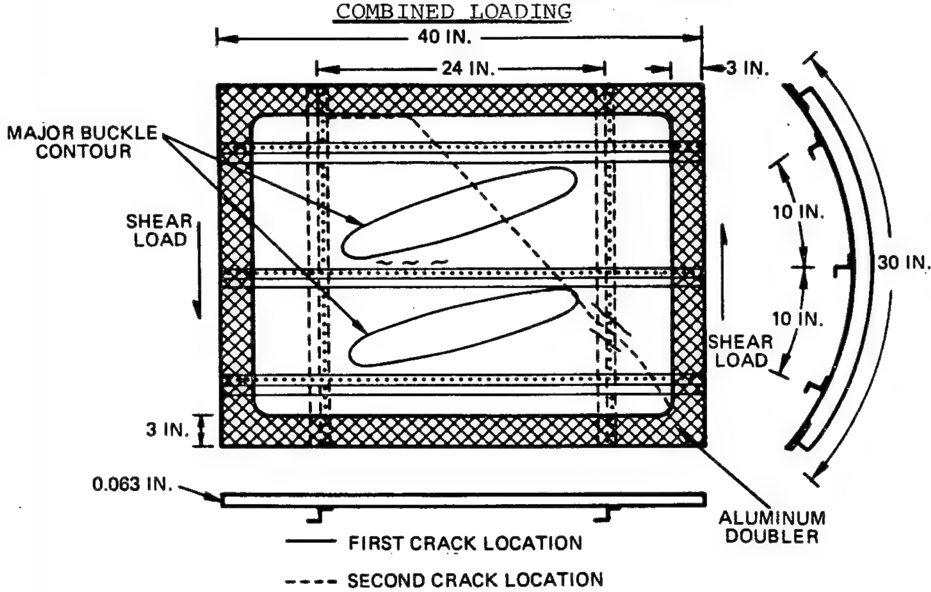


MAX SPECTRUM LOAD = 71.6% OF FAILURE LOAD

FATIGUE RESPONSE OF FLAT PANELS UNDER COMBINED LOADING



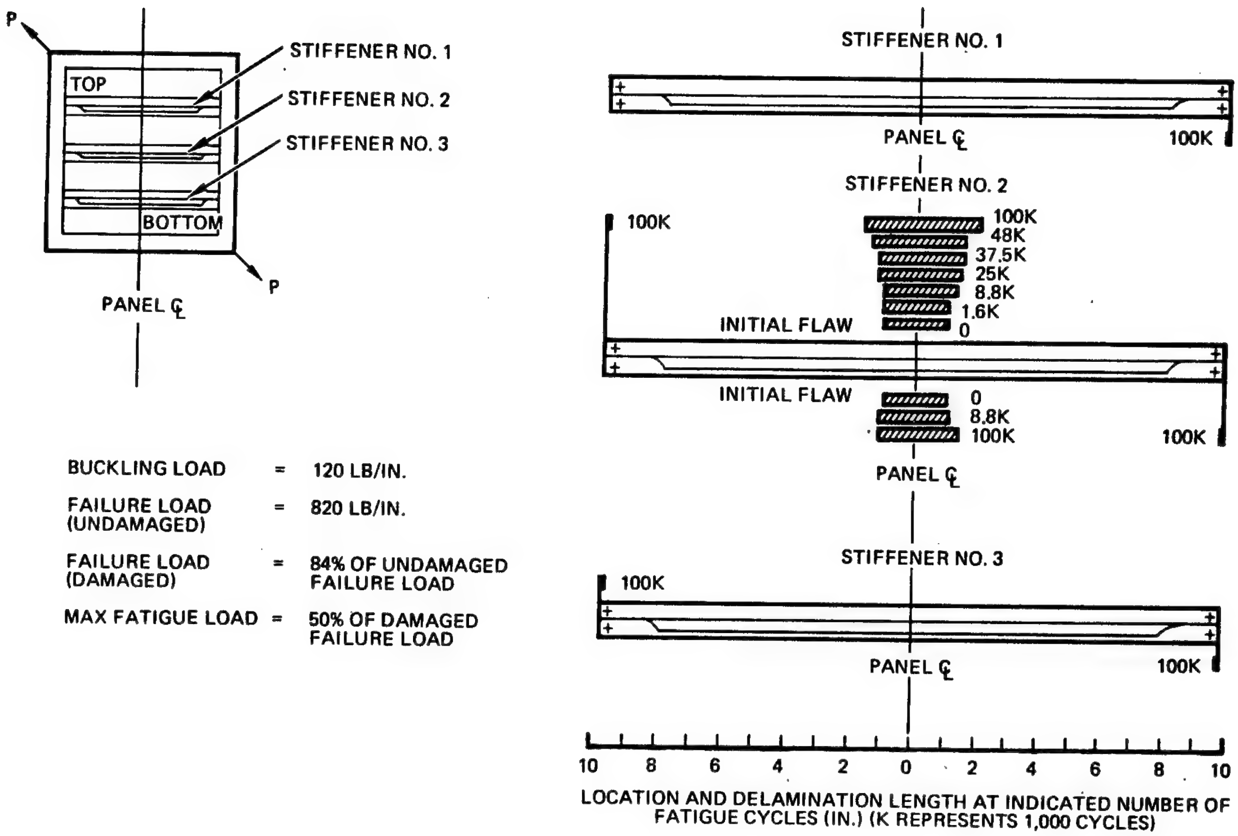
METAL SHEAR PANEL FATIGUE DATA



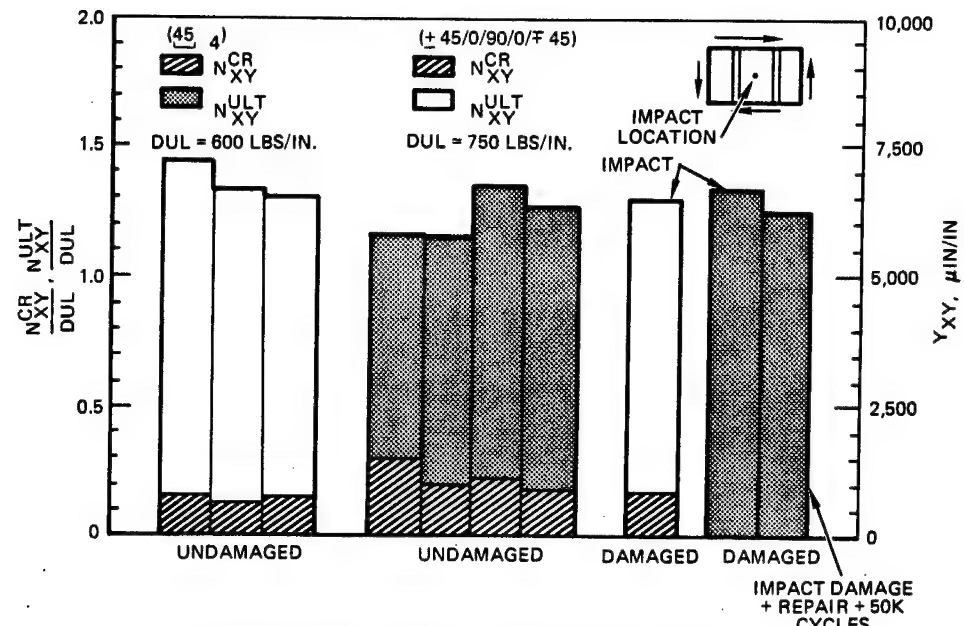
FATIGUE CRACKS IN METAL SHEAR PANEL

FATIGUE CRACKS IN METAL COMPRESSION PANEL

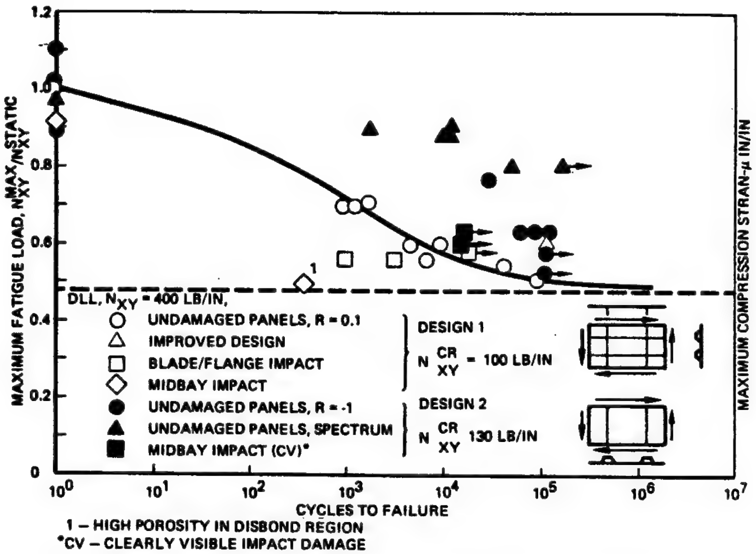
DURABILITY OF POSTBUCKLED PANELS



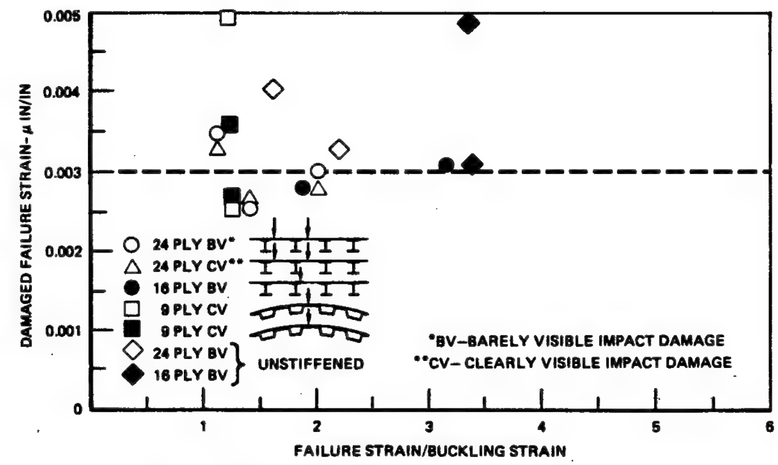
INFLUENCE OF SKIN-STIFFENER DISBOND



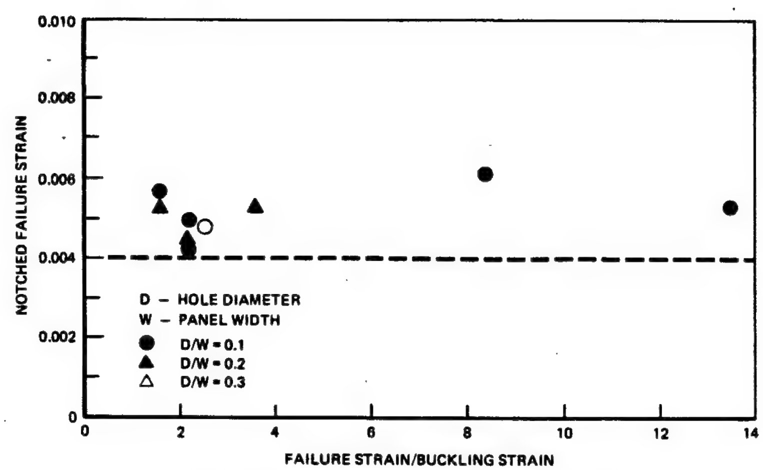
INFLUENCE OF IMPACT DAMAGE ON SHEAR PANEL STATIC STRENGTH



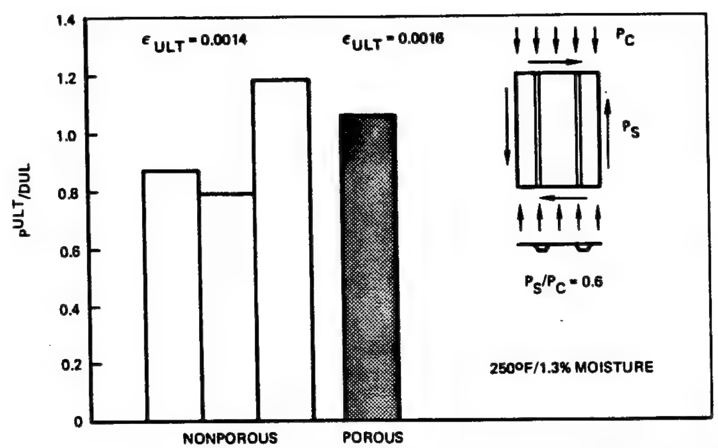
INFLUENCE OF IMPACT ON COMPOSITE SHEAR PANEL FATIGUE RESPONSE



INFLUENCE OF IMPACT DAMAGE ON COMPOSITE COMPRESSION



INFLUENCE OF HOLES ON COMPOSITE COMPRESSION PANELS

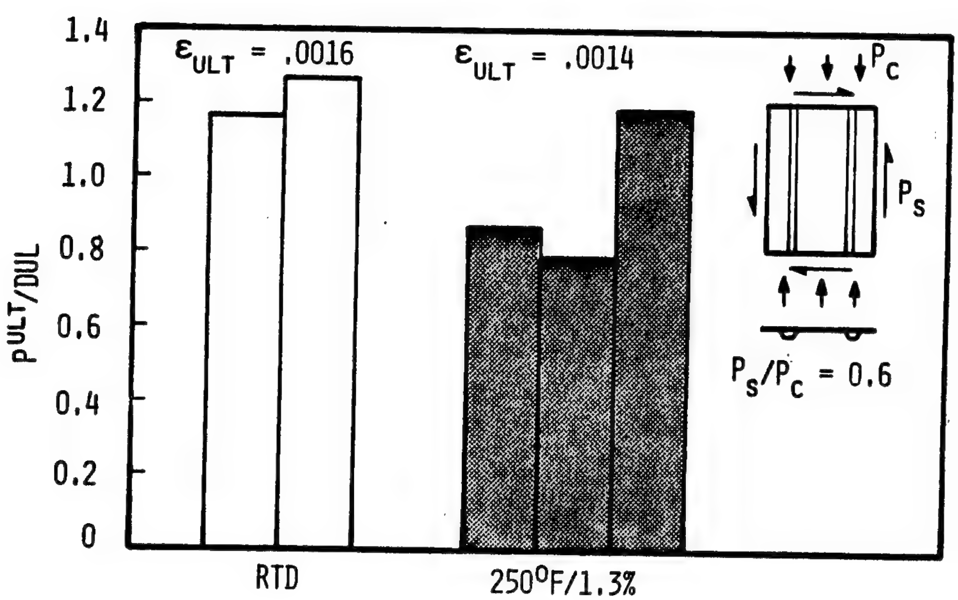


INFLUENCE OF POROSITY ON FLAT COMPOSITE PANELS UNDER COMBINED LOADING

INFLUENCE OF ENVIRONMENT ON COMPOSITE PANELS

- ENVIRONMENTAL EFFECTS ON PANEL STIFFNESS INSIGNIFICANT
- AVAILABLE ENVIRONMENTAL DATA SUFFICIENT TO EVALUATE STRENGTH REDUCTION
  - -- NORTHROP DATA
    - IR&D
    - AIR FORCE SPONSORED WING/FUSELAGE CONTRACT
  - -- INDUSTRY DATA
    - TWO ONGOING NAVY PROGRAMS

INFLUENCE OF ENVIRONMENT ON STATIC STRENGTH



REPAIR TECHNIQUES

- AVAILABLE DATA INDICATE THAT REPAIRS OF COMPOSITE PANELS FOR COMMONLY OCCURRING IMPACT DAMAGE AND DISBONDS ARE NOT NEEDED

- REPAIR TECHNIQUES FOR BUCKLING RESISTANT STRUCTURES CAN BE USED TO REPAIR POSTBUCKLED PANELS

  -- SCARFED PATCH FOR COMPOSITE PANELS

  -- BOLTED DOUBLER FOR METAL PANELS

  -- COMPOSITE PATCH FOR METAL PANELS

81-03242

FAILURE MODE OF REPAIRED COMPRESSION PANEL

COMBINED LOADING PANEL AFTER REPAIR

POSTBUCKLING CHARACTERISTICS OF STIFFENED GRAPHITE-EPOXY SHEAR WEBS

Marshall Rouse

Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665

ABSTRACT

Structurally efficient fuselage panels are often designed to allow buckling to occur at applied loads below ultimate. Recent interest in applying graphite-epoxy materials to fuselage primary structures has lead to several studies of postbuckling behavior of graphite-epoxy structural components. NASA has been studying the postbuckling behavior of composite panels for potential transport fuselage applications.

The present paper presents some recent results of a study of the postbuckling behavior of flat stiffened graphite-epoxy shear webs loaded into the postbuckling range. The response and failure characteristics of specimens studied experimentally will be described. Results will be presented for 19-, 25-, and 33-ply shear webs. All of the shear webs had 22.5-inch by 22.5-inch test sections. The 19-ply shear webs had a 4.5-inch stiffener spacing and the 25- and 33-ply webs had a 7.5-inch stiffener spacing. There was a common stiffener design for all of the webs that were tested. The stiffeners were attached to the skins by mechanical fasteners and/or adhesive bonding. Also, some of the specimens had stiffeners attached using a modified mechanical fastener concept in an attempt to improve the performance of the specimens loaded into the postbuckling range.

The present paper will present results which show the influence of skin thickness and stiffener spacing on the postbuckling response of graphite-epoxy shear webs. Results will also be presented that describe the effect on failure of different attachment concepts.

POSTBUCKLING CHARACTERISTICS OF STIFFENED GRAPHITE- EPOXY SHEAR WEBS

MARSHALL ROUSE

NASA LANGLEY RESEARCH CENTER
HAMPTON, VIRGINIA

OBJECTIVE

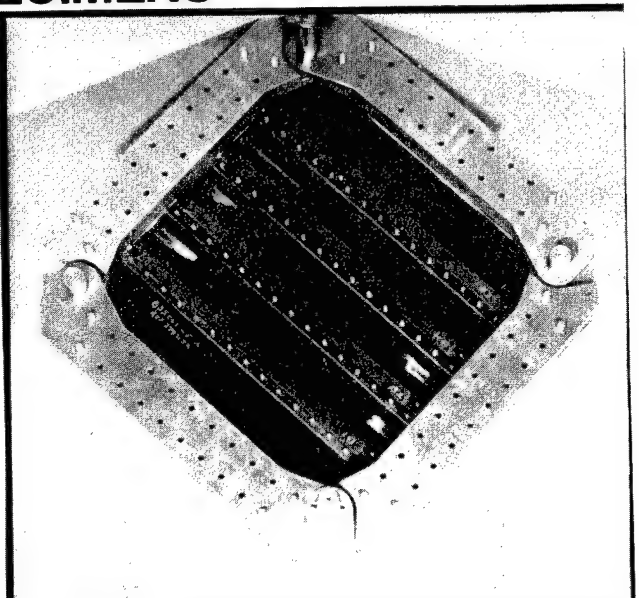
O DESCRIBE THE POSTBUCKLING BEHAVIOR OF STIFFENED GRAPHITE-EPOXY SHEAR WEBS

- o DIFFERENT SKIN THICKNESS
- o DIFFERENT STIFFENER SPACING
- o DIFFERENT STIFFENER ATTACHMENT CONCEPTS

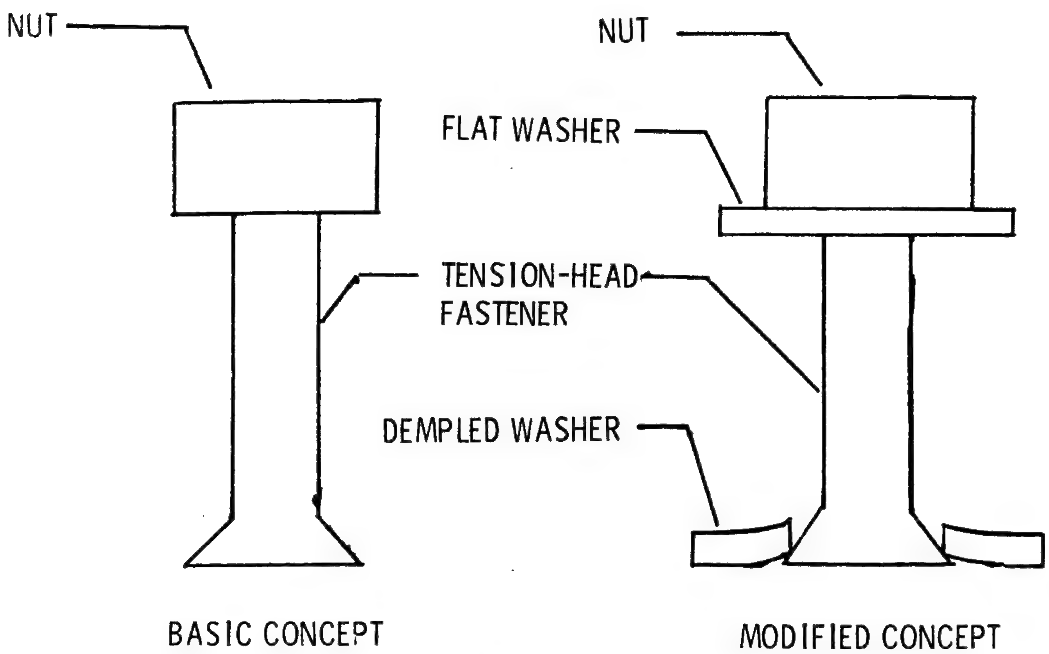
O DESCRIBE THE FAILURE CHARACTERISTICS OF STIFFENED GRAPHITE-EPOXY SHEAR WEBS LOADED INTO THE POSTBUCKLING RANGE

TEST SPECIMENS

- AS4/3502 Graphite-Epoxy Tape
- 4 Stiffeners With 4.5 in Spacing Skin Laminates $[\pm 45, \mp 45, \pm 45, 90_2, \overline{0}_3]_s$
- 2 Stiffeners With 7.5 in Spacing Skin Lamintes
  $[(\pm 45, \mp 45)_2, 90_2, \overline{0}_5]_s$
  $[(\pm 45, \mp 45)_3, 0_2, \overline{90}_5]_s$

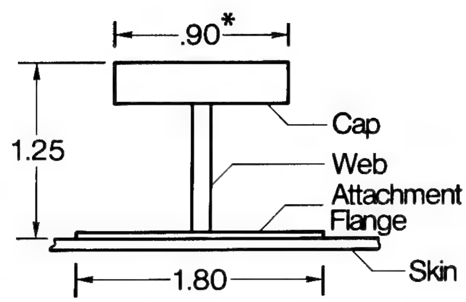


DETAILS OF MECHANICAL FASTENERS

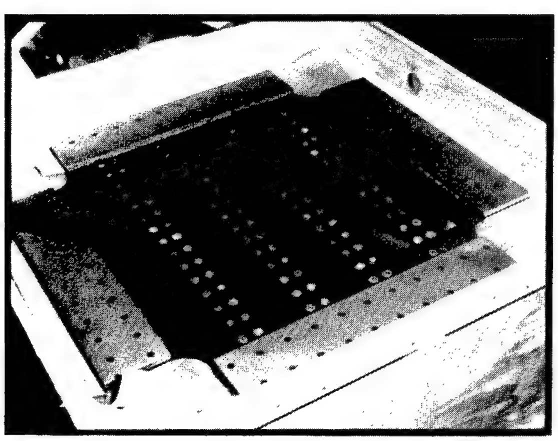


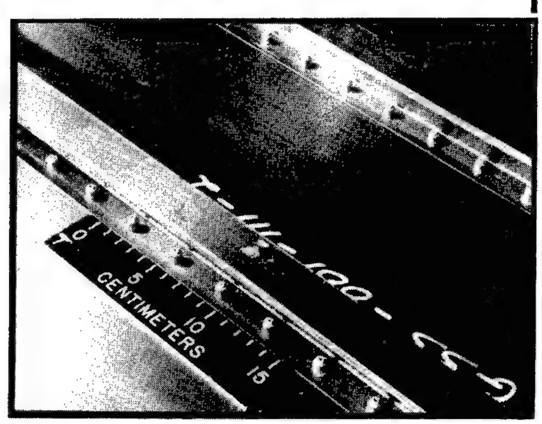
STIFFENER CONFIGURATIONS

- Common Stiffener for Each Skin Thickness
  $[\pm 45, \mp 45, 0_2, \pm 45, 0_6, 90, 0_2]_s$ Cap
  $[\pm 45, \mp 45, 0_2 / \pm 45]_s$ Web
  $[\mp 45, 0_2, \pm 45, \mp 45]$ Attachment Flange
- Secondarily Bonded and/or Mechanically Fastened

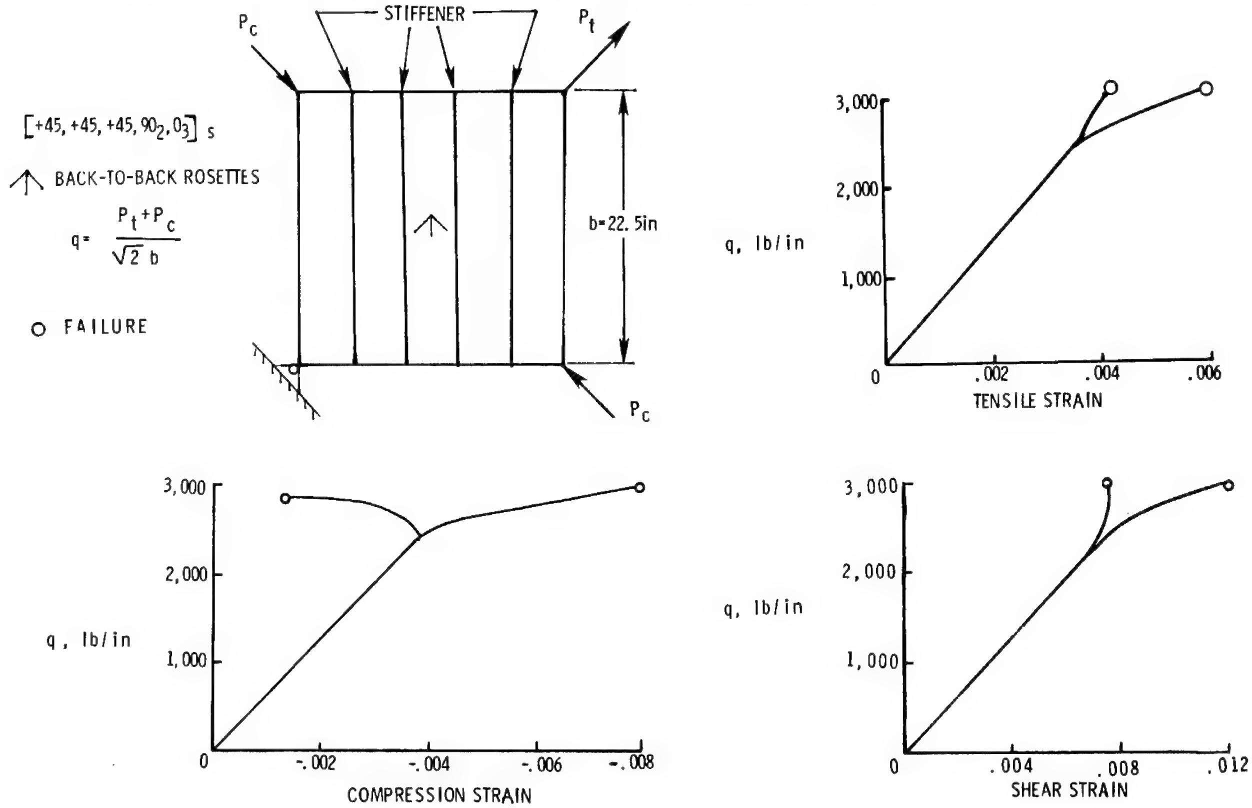


*Except 2-Stiffer Web which has Cap Width of 1.20 in.

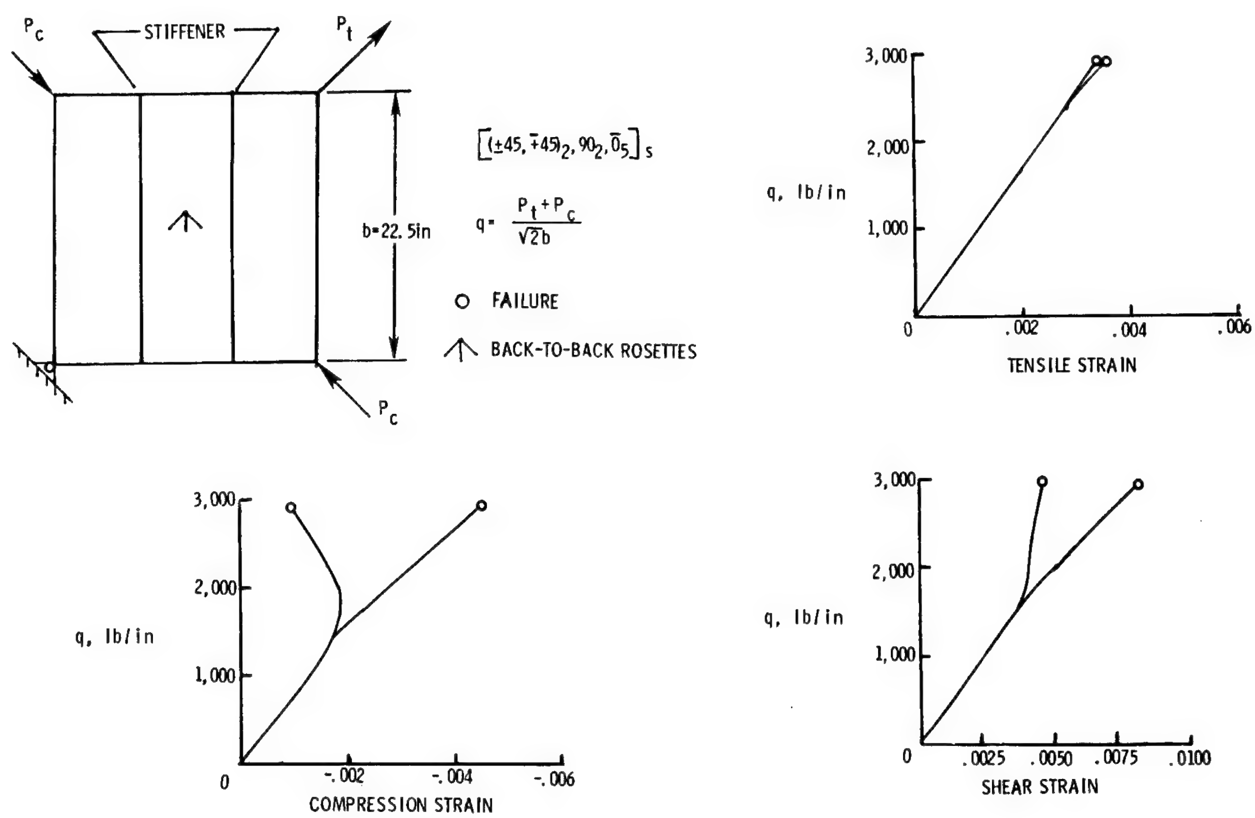




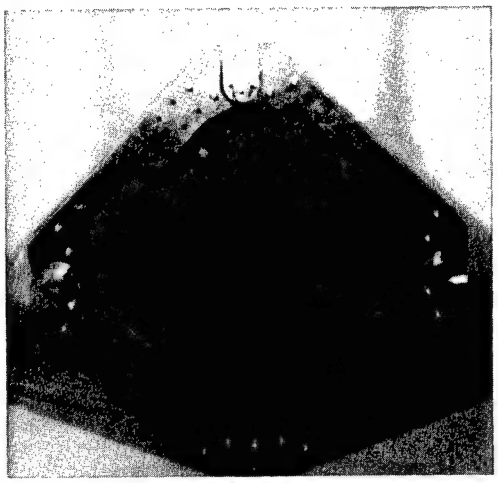
TYPICAL EXPERIMENTAL RESULTS FOR 19-PLY SHEAR WEB

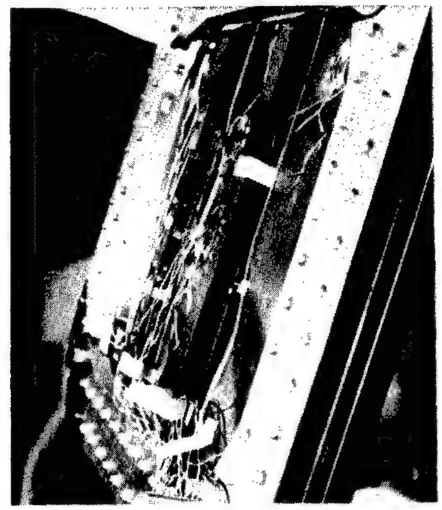


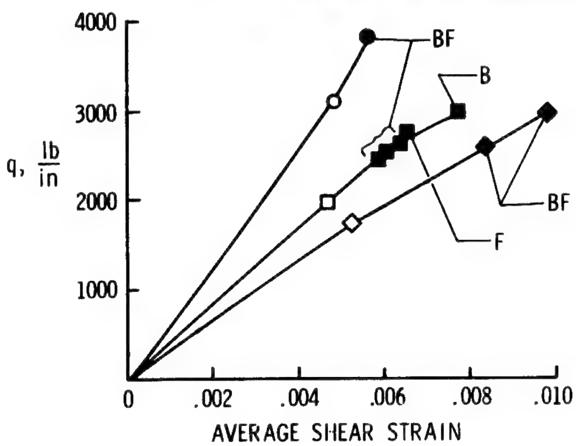
TYPICAL EXPERIMENTAL RESULTS FOR 25-PLY SHEAR WEB



POSTBUCKLING BEHAVIOR OF STIFFENED GRAPHITE-EPOXY SHEAR WEBS







B - BONDED ONLY
F - MECHANICAL FASTENERS ONLY
BF - BONDED AND MECHANICAL FASTENERS

BUCKLING		FAILURE
○	$[(\pm45, \mp45)_3, 90_2, \bar{0}_5]_s$	●
□	$[(\pm45, \mp45)_2, 0_2, \overline{90}_5]_s$	■
◇	$[\pm45, \mp45, \pm45, 90_2, \bar{0}_3]_s$	◆

CONCLUDING REMARKS

O SHEAR WEBS WITH BONDED STIFFENERS FAILED DUE TO SKIN- STIFFENER SEPARATION

O FAILURE CHARACTERISTICS ARE INFLUENCED BY METHOD OF STIFFENER ATTACHMENT

O FASTENERS PULLED THROUGH ATTACHMENT FLANGE AND/OR SKIN

THE MECHANICS OF FLAW GROWTH AND FRACTURE OF FIBER COMPOSITES AND ADHESIVE JOINTS

Professor S. S. Wang

Department of Theoretical and Applied Mechanics
University of Illinois
Urbana, IL 61801
(217) 333-1835

Abstract

This research is concerned with the fundamental mechanics of flaw growth and fracture in fiber-reinforced composite materials and adhesively bonded joints used in high-performance engineering structures and components such as modern aircraft and high-speed hydrofoil craft constructions. Research efforts have been concentrated on developments of analytical methods for studying the flaw behavior and failure response of composite materials and structures under various loading modes including impact and compression. Both advanced continuous-fiber composite laminates and short-fiber composites are considered. In the area of fracture of adhesive joints, three classes of commonly used adhesive joints are studied: metal-to-metal adhesive joints, metal-to-fiber-composite joints, and fiber-composite-to-fiber composite joints. Emphasis of this research program are placed on material and geometric nonlinear mechanics aspects of fracture in composites and adhesive joints. Interactions between flaw growth, fracture and structural instability are also investigated. Detail information on the basic mechanics and mechanism of crack initiation, growth and fracture in fiber composites and bonded structures will be given in the lecture.

THE EFFECT OF CONSTITUENT PROPERTIES ON COMPRESSIVE FAILURE MECHANISMS IN UNIDIRECTIONAL COMPOSITES

H. T. Hahn
Washington University
St. Louis, MO

and

Jerry G. Williams
NASA Langley Research Center
Hampton, VA

ABSTRACT

The objectives of this research project are to identify compressive failure mechanisms in unidirectional composites, to define the effect of matrix and fiber properties on compressive behavior, and to develop an analytical model to predict unidirectional compressive strength based on observed failure mechanisms.

Failure in unidirectional laminates under compressive loading has been found to be strongly dependent upon the stability provided by the matrix to the slinder high modulus fiber (ref. 1). Although delamination between fibers often occurs, it is commonly preceeded by buckling related failure of the fibers. Two stability related failure modes have been observed to occur: kinking and microbuckling. Kinking failure is characterized by the formation of a kink band in which two fractures per fiber occur and the short broken fibers align in a regular parallel pattern rotated at an angle relative to the initial fiber direction. Microbuckling type failure occurs when the fibers remain in the postbuckled state following failure and the fiber typically exhibits multiple fractures in it's highly deformed state. In stability related failure modes the fiber fracture surface is orthogonal to the axis of the fiber.

Two types of specimens were used in the study including a fiber bundle embedded in a transparent epoxy block and the unidirectional IITRI specimen. The transparent epoxy permitted in situ observation using a microscope of the failure of fibers in the bundle. Failed specimens were also examined using optical and scanning electron microscopy to determine failure initiation and propagation mechanisms.

Fibers used in embedded bundle specimens included: Owens Corning E-glass and Union Carbide T300, T700, and T75 graphite. Each type of fiber was combined with two resin systems: Shell Epon 815/V140 (60%/40% mixture by weight) and Shell Epon 828/Z (80%/20% mixture by weight). The extensional and shear moduli for Epon 828/Z are higher than the corresponding properties for Epon 815/V140. In both resin systems the failure mode of the E-glass, T300 and T700 bundle specimens was found to be fiber microbuckling. The higher modulus T75 graphite bundle specimens failed without evidence of loss of stability in a classical fiber compression failure mode in which the fiber failure surface is oriented at a nonorthogonal angle relative to the axis of the fiber. Microscopic examination revealed that the failure of individual T75 fibers, however, is due to kinking of the fibrils which constitute the fiber. Fibers buckled at lower strains in the lower modulus Epon 815/V140 resin than in the stiffer Epon 828/Z resin. The buckling strains and the average lengths of broken segments followed the trend predicted by analysis (ref. 2) for a single fiber embedded in an infinite matrix.

Compressive behavior of unidirectional laminates using the IITRI specimen was studied using two different graphite fibers: T300 and T700, and four different epoxy resins: Narmco 5208, American Cyanamid BP907, and Union Carbide 4901/MDA and 4901/mPDA. The predominant failure mode in the unidirectional laminate specimens on both the macroscopic (specimen thickness) and microscopic (individual fibers) scales was initiated due to the stability failure of the fibers. On the microscopic scale, the stability failure for BP907 resin specimens typically was microbuckling. For higher modulus resin laminates the microscopic scale failure was typically the formation of a kink band. Failure was usually observed to initiate at a free edge; probably because there is less support to stabilize the fibers in this region and grew inward at an oblique angle to the fibers. Compressive strength, and compressive modulus to a lesser extent, of unidirectional composites increased with increasing resin tensile modulus. The strains at which failure occurred in IITRI specimens were lower than the corresponding compressive failure strains for fiber bundle specimens.

An analytical model was developed to predict the compressive strength of a unidirectional laminate. The model assumes that compressive failure of the composite is a result of loss of fiber stability in a classical shear mode (ref. 3) and the fibers fail due to high stresses which develop in the postbuckled state. Matrix material nonlinearity is introduced through the use of the secant shear modulus to account for nonlinear stress-strain behavior. A simplified calculation for the distributed moments and forces on a fiber introduces a relationship for compressive strength involving the fiber volume fraction. Available experimental data (refs. 4,5) correlates very well with the analytical compressive strength predicted by the new model.

Identification of commercial products in this report is used to describe adequately the test materials. Neither the identification of these commercial products nor the results of the investigation published herein constitute official endorsement, expressed or implied, of any such product by NASA or Washington University.

REFERENCES

1. Hahn, H. T.; and Williams, Jerry G.: Compression Failure Mechanisms in Unidirectional Composites. Presented at the Composite Materials: Testing and Design (7th Conference), Philadelphia, PA, April 1984. NASA TM-85834, August 1984.

2. Lanir, Y.; and Fung, Y. C. B.: Fiber Composite Colums Under Compression. J. Comp. Mat., Vol. 6, 1972, pp.387-401.

3. Rosen, V. W.: Mechanics of Composite Strengthening in Fiber Composite Materials. ASM, 1965, pp. 37-75.

4. Palmer, R. J.: Investigation of the Effect of Resin Material on Impact Damage to Graphite-Epoxy Composites. NASA CR-165677. March, 1981.

5. Williams, Jerry G.; and Rhodes, Marvin D.: The Effect of Resin On the Impact Damage Tolerance of Graphite-Epoxy Laminates. Composite Materials: Testing and Design (6th Conference), ASTM STP 787, I. M. Daniel, ed., 1982, pp. 450-480.

THE EFFECT OF CONSTITUENT PROPERTIES ON COMPRESSIVE FAILURE MECHANISMS IN UNIDIRECTIONAL COMPOSITES

H.T. HAHN (*)
WASHINGTON UNIVERSITY
ST. LOUIS, MO.

JERRY G. WILLIAMS
NASA LANGLEY RESEARCH CENTER
HAMPTON, VA.

Tenth Annual Mechanics of Composites Review
Dayton, Ohio
October 15-17, 1984

* SUPPORTED BY NASA LANGLEY RESEARCH CENTER
UNDER GRANT NAG -1-295

OBJECTIVES

- To Identify Compressive Failure Mechanisms in Unidirectional Composites
- To Determine the Effect of Matrix and Fiber Properties on Compression Performance
- To Develop an Analytical Model to Predict Compression Strength

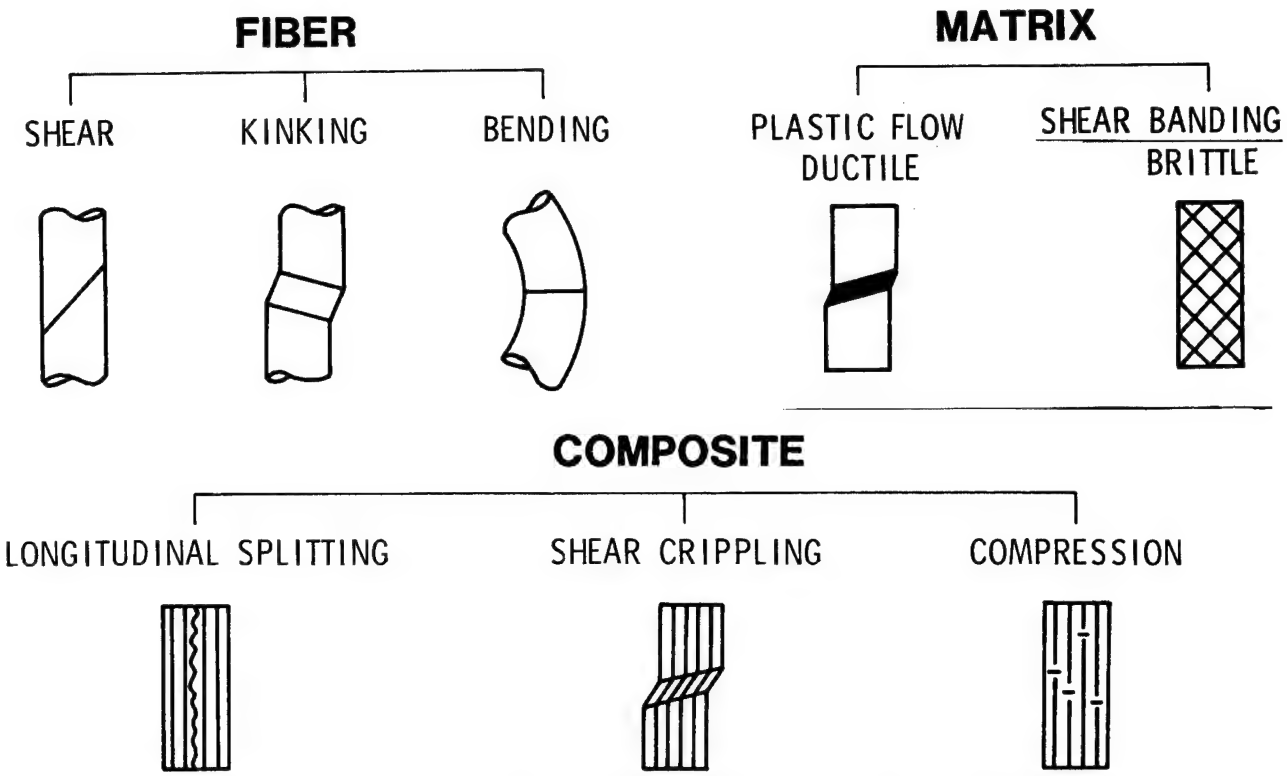
MATERIALS USED IN BUNDLE SPECIMENS

	MATERIAL	DIAMETER, μm	NO. OF FILAMENTS	MODULUS, GPa	TENSION STRENGTH, MPa	FAILURE STRAIN, %
FIBER	E-GL	13.5	200	72.3	3450	4.80
	T700	5.1	4500	230.0	4206	1.83
	T300	7.0	3000	234.0	3310	1.34
	T75	9.7	2000	517.0	2068	0.40
RESIN	EPON 828/ Z(80/20)	-	-	3.45	85.4	9
	EPON 815/ V140(60/40)	-	-	2.13	45.5	14

MATERIALS USED IN COMPOSITE SPECIMENS

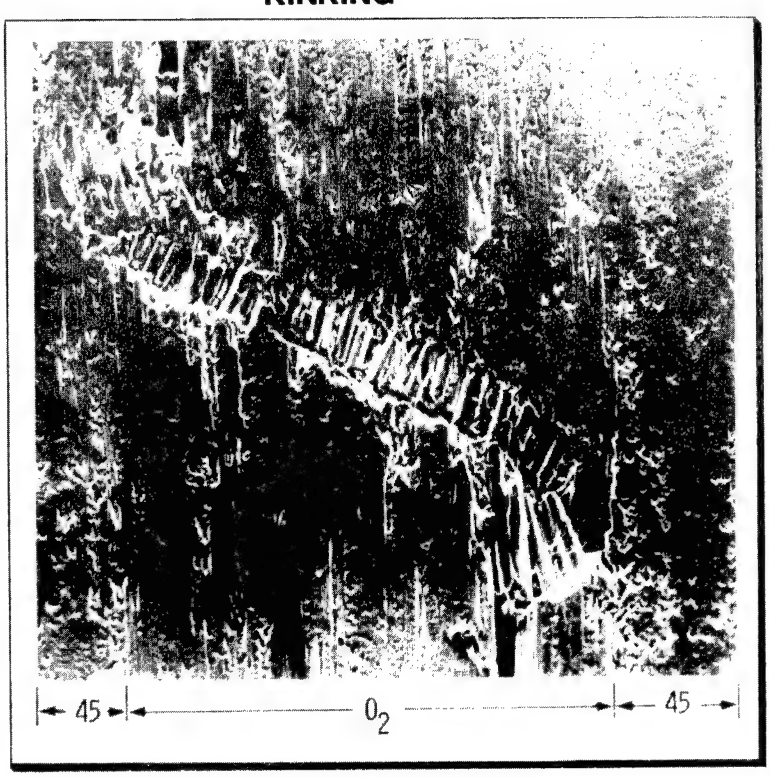
	MATERIAL	MANUFACTURER	TENSION MODULUS, GPa	TENSION FAILURE STRESS, MPa	TENSION FAILURE STRAIN, %
FIBER	T300	UNION CARBIDE	230	3100	0.012
	T700	UNION CARBIDE	248	4550	0.018
EPOXY	5208	NARMCO	4.0	57.2	1.8
	BP907	AMERICAN CYANAMID	3.1	89.5	4.8
	4901/MDA	UNION CARBIDE	4.62	103.4	4.0
	4901/mPDA	UNION CARBIDE	5.46	115.1	2.4

FAILURE MODES IN COMPRESSION



FIBER STABILITY INITIATED FAILURE MODES

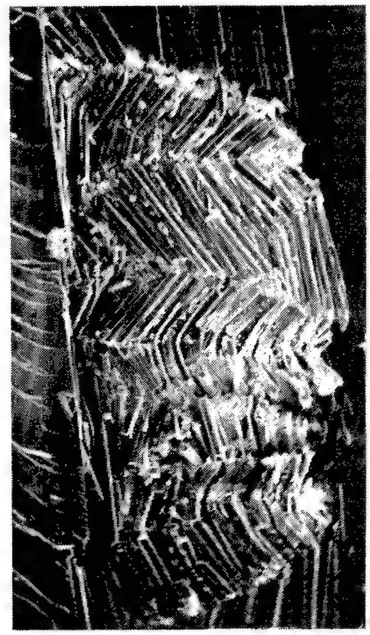
KINKING



MICROBUCKLING

MICROBUCKLING OF E-GLASS BUNDLES

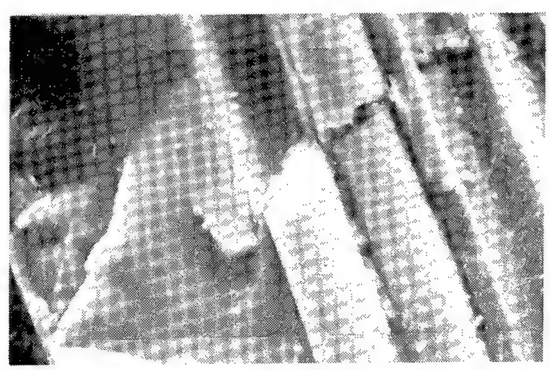
WEAK EPON 815



EPON 815
(SIMILAR IN EPON 828)

FAILURE MODES OF GRAPHITE BUNDLES

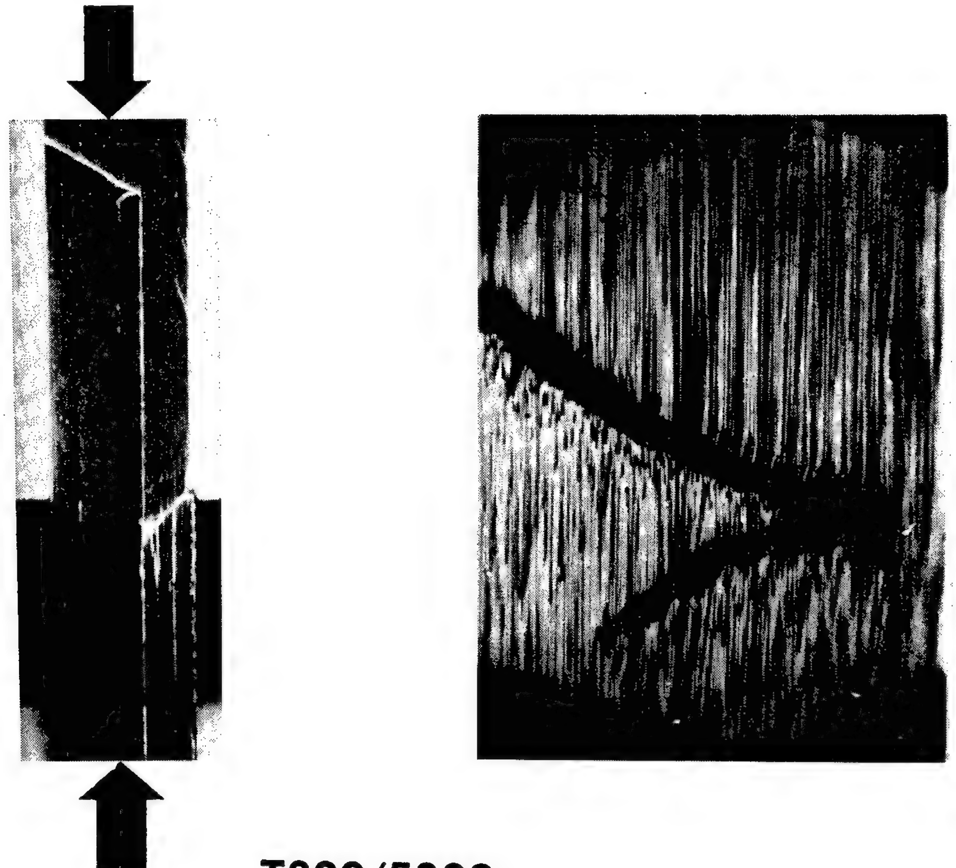
KINKING IN T700/EPON 815
(SIMILAR IN T700/EPON 828,
T300/EPON 815,
AND T300/EPON 828)



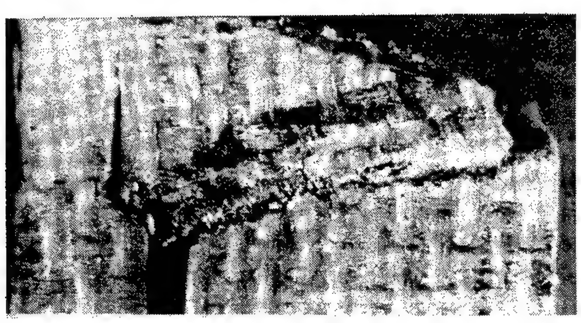
SHEAR FAILURE IN T75/EPON 815
(SIMILAR IN T75/EPON 828)

FIBER KINKING IN GRAPHITE/EPOXY COMPOSITES

UNIDIRECTIONAL LAMINATE



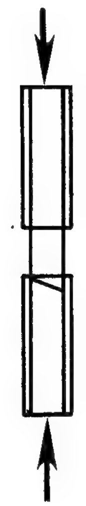
T300/5208

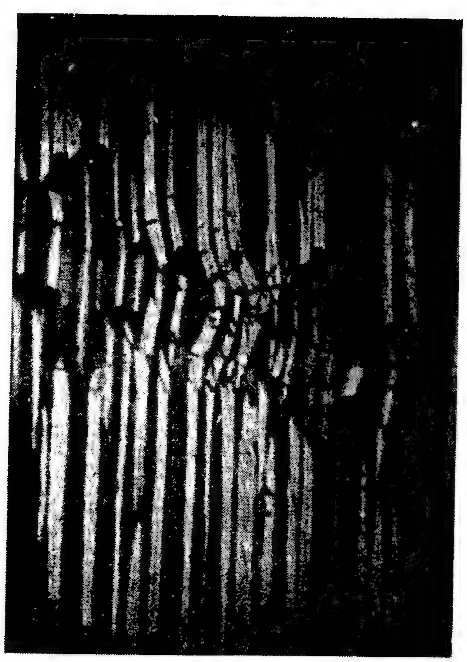


T300/4901/MDA

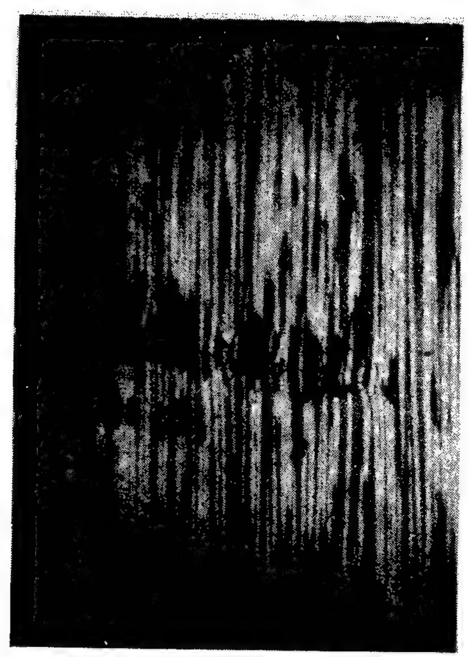
FIBER MICROBUCKLING IN T300/BP907

UNIDIRECTIONAL LAMINATE



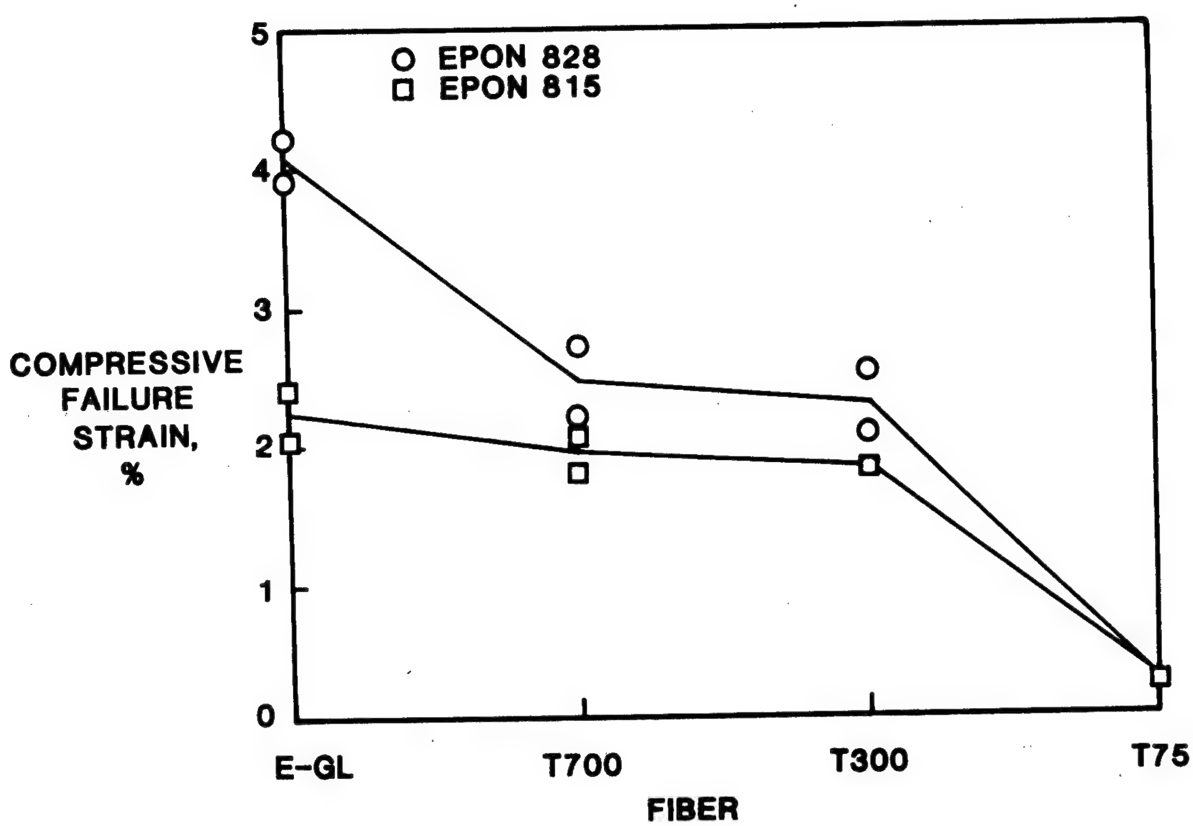


MICROBUCKLING OF FIBERS



TIP OF MICROBUCKLING BAND

COMPRESSIVE FAILURE STRAINS OF BUNDLES



ANALYTICAL FAILURE MODEL

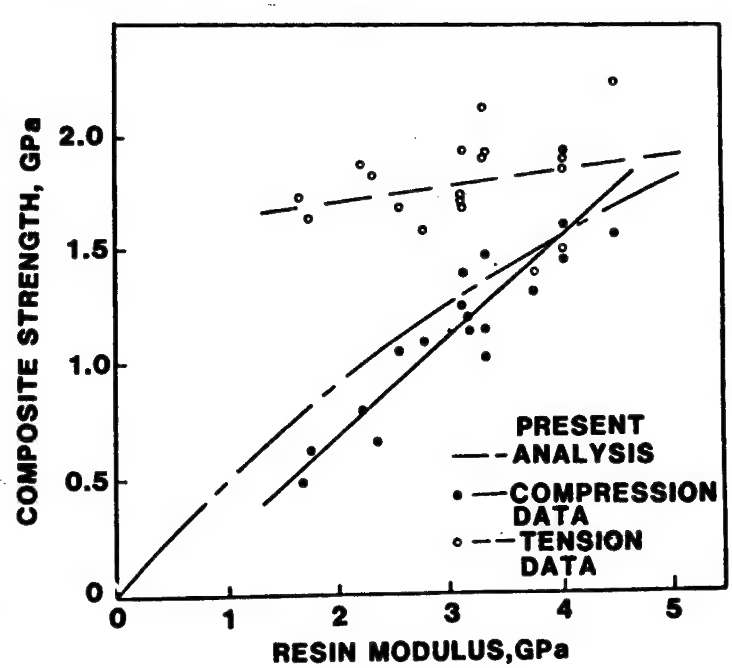
$$\sigma_c = V_f G_{LT} \frac{1}{1 + (\pi f_0/\ell)/\gamma_{LT}}$$

INITIAL DEFLECTION :

$$v_0 = f_0 \cos \frac{\pi}{\ell} x$$

G_{LT} : SECANT MODULUS

γ_{LT} : FAILURE STRAIN IN SHEAR



DATA FROM NASA CR 165677

CONCLUSIONS

FIBER BUNDLE SPECIMENS

- o A FIBER BUNDLE EMBEDDED IN RESIN IS EFFECTIVE SPECIMEN FOR STUDYING COMPRESSIVE FAILURE.
- o COMPRESSIVE STRENGTH CAPABILITIES OF FIBERS ARE NOT FULLY UTILIZED IN COMPOSITES.
- o UNDER COMPRESSION, T75 FIBERS FAIL IN SHEAR BY KINKING OF FIBRILS WHILE T300 AND T700 FIBERS FAIL BY FIBER MICROBUCKLING OR KINKING. MICROBUCKLING IS MOST PRONOUNCED WITH E-GLASS FIBERS.
- o MICROBUCKLING OF A FIBER BUNDLE CAN BE GLOBALLY DISTRIBUTED FOR A SOFT RESIN; HOWEVER, IT IS USUALLY LOCALIZED FOR A STIFF RESIN.

UNIDIRECTIONAL COMPOSITES

- o AN IMPROVED ANALYTICAL MODEL FOR COMPRESSIVE STRENGTH HAS BEEN DEVELOPED.
- o FIBER FAILURE FOR BP907 RESIN IS MICROBUCKLING; IN STIFFER RESINS, HOWEVER, IT IS KINKING.
- o COMPRESSIVE STRENGTH INCREASES WITH RESIN MODULUS.
- o LONGITUDINAL SPLITTING IS SUPPRESSED BY TOUGH RESINS.
- o LAMINATE COMPRESSIVE STRENGTH AND TOUGHNESS CAN BE IMPROVED USING ADVANCED RESINS AND FIBERS.

AN IMPROVED THEORY FOR LAMINATED COMPOSITE BEAMS

Manuel Stein and Dawn C. Jegley
Structural Mechanics Branch
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665

ABSTRACT

Classical beam bending theory predicts deformations and longitudinal stresses which are comparable to the actual deformations and longitudinal stresses given by three dimensional elasticity for thin beams of homogenous material. Conventional transverse shearing theory makes similar predictions for sandwich construction and for thicker beams of homogenous material. An average value of the transverse shear stress may be found from the deformations using the strain-displacement relation and conventional transverse shearing theory, but reasonable distributions of transverse shear stress and transverse direct stress can only be found for isotropic beams in a special solution that uses beam results and elasticity theory. No satisfactory approach is available to obtain transverse stresses for a nonhomogeneous material such as laminated composites. The idea of determining accurately the transverse shear stress and transverse direct stress in addition to the displacements and longitudinal stress is important for beams even though the transverse stresses are small compared to the longitudinal stress. These transverse stresses are important when the structures are relatively weak in the transverse direction and when the loads are sensitive to the transverse stiffness.

A new approach for predicting the stresses in laminated composite beams is presented. In the present theory nonlinear strains for three dimensional elasticity are determined. Trigonometric series representation for the unknowns in terms of the variable through-the-thickness is set up. A variational procedure is applied using the potential energy method so that a few terms in the series can be used to get accurate displacements and stresses. The terms in the series are chosen such that by including only the first few terms, the series can reduce to conventional transverse shearing (Timoshenko) theory and for thin beams to classical (Kirchhoff) theory.

Two loading cases are considered. First, the present theory is compared to the corresponding elasticity solution for the case of a sinusoidal pressure loading on a simply supported beam. This loading case is applied to isotropic and laminated composite beams. Material properties of aluminum and a three layer $(0/90/0)_T$ laminated composite beam are used. Second, the loading case of a midspan concentrated load applied to a simply supported beam is studied. Properties of $(0/90/0)_T$ and unidirectional beams are used. These results are compared to experimental data for the unidirectional laminated composite beam.

Using the potential energy method, the present theory accurately predicts stresses and displacements in isotropic and unidirectional composite beams and the longitudinal stress and displacements in beams with layers of differing material properties. The present theory may provide a useful way of interpreting experimental data to obtain an accurate value of the through-the-thickness shear modulus.

AN IMPROVED THEORY FOR LAMINATED COMPOSITE BEAMS

M. Stein

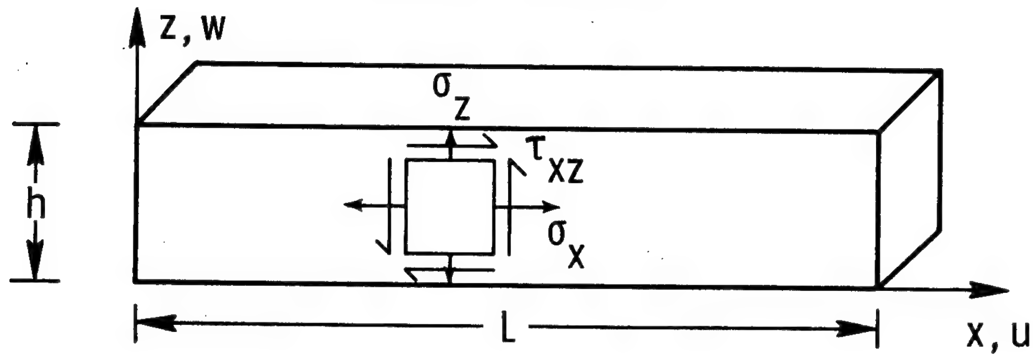
D. C. Jegley

NASA Langley Research Center

OBJECTIVE

To develop accurate multi-layered beam theory from which transverse shear and normal stresses can be determined

BEAM MODEL



INFINITE SET

$$u = u_a(x)\left[\frac{z}{h} - \frac{1}{2}\right] + \sum_{n=1,3,5}^{\infty} u_n(x)\cos\frac{n\pi z}{h}$$

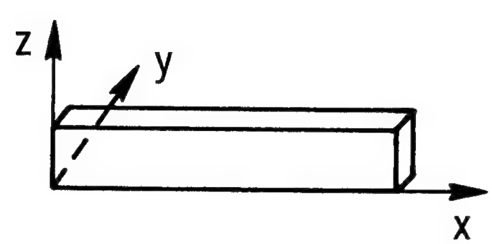
$$w = w_o(x) + w_a(x)\left[\frac{z}{h} - \frac{1}{2}\right]^2 + \sum_{n=1,3,5}^{\infty} w_n(x)\sin\frac{n\pi z}{h}$$

SOLUTION APPROACHES

- Potential energy
- Complementary energy
- Reissner energy

METHOD

- 3-dimensional elasticity with $\sigma_y = 0$ and stresses and strains independent of y



POTENTIAL ENERGY METHOD USING VIRTUAL WORK

$$\delta \pi = \int_0^L \int_0^h \left(\sigma_x \delta \varepsilon_x + \sigma_z \delta \varepsilon_z + \tau_{xz} \delta \gamma_{xz}\right) dzdx = 0$$

STRAIN-DISPLACEMENT AND STRESS-STRAIN RELATIONS

$$\varepsilon_x = \frac{\partial u}{\partial x} \qquad \sigma_x = E \varepsilon_x$$

$$\varepsilon_z = \frac{\partial w}{\partial z} \qquad \sigma_z = E_z \varepsilon_z$$

$$\gamma_{xz} = \frac{\partial u}{\partial z} + \frac{\partial w}{\partial x} \qquad \tau_{xz} = G \gamma_{xz}$$

ASSUMED DISPLACEMENTS

CLASSICAL THEORY:

$$u(x,z) = -\frac{dw_0}{dx}(x)\left(\frac{z}{h} - \frac{1}{2}\right)h$$

$$w(x,z) = w_0(x)$$

TIMOSHENKO THEORY:

$$u(x,z) = u_a(x)\left(\frac{z}{h} - \frac{1}{2}\right)$$

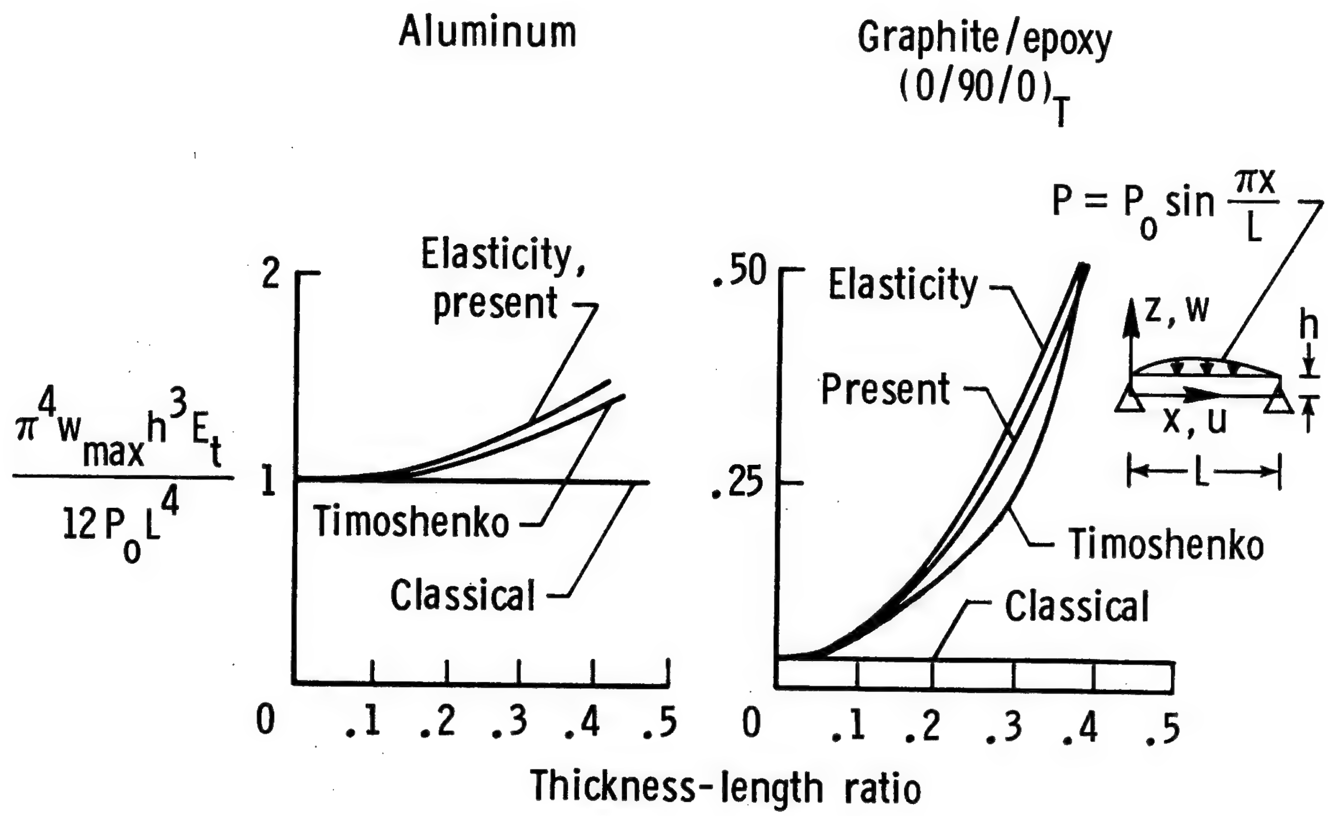
$$w(x,z) = w_0(x)$$

PRESENT THEORY:

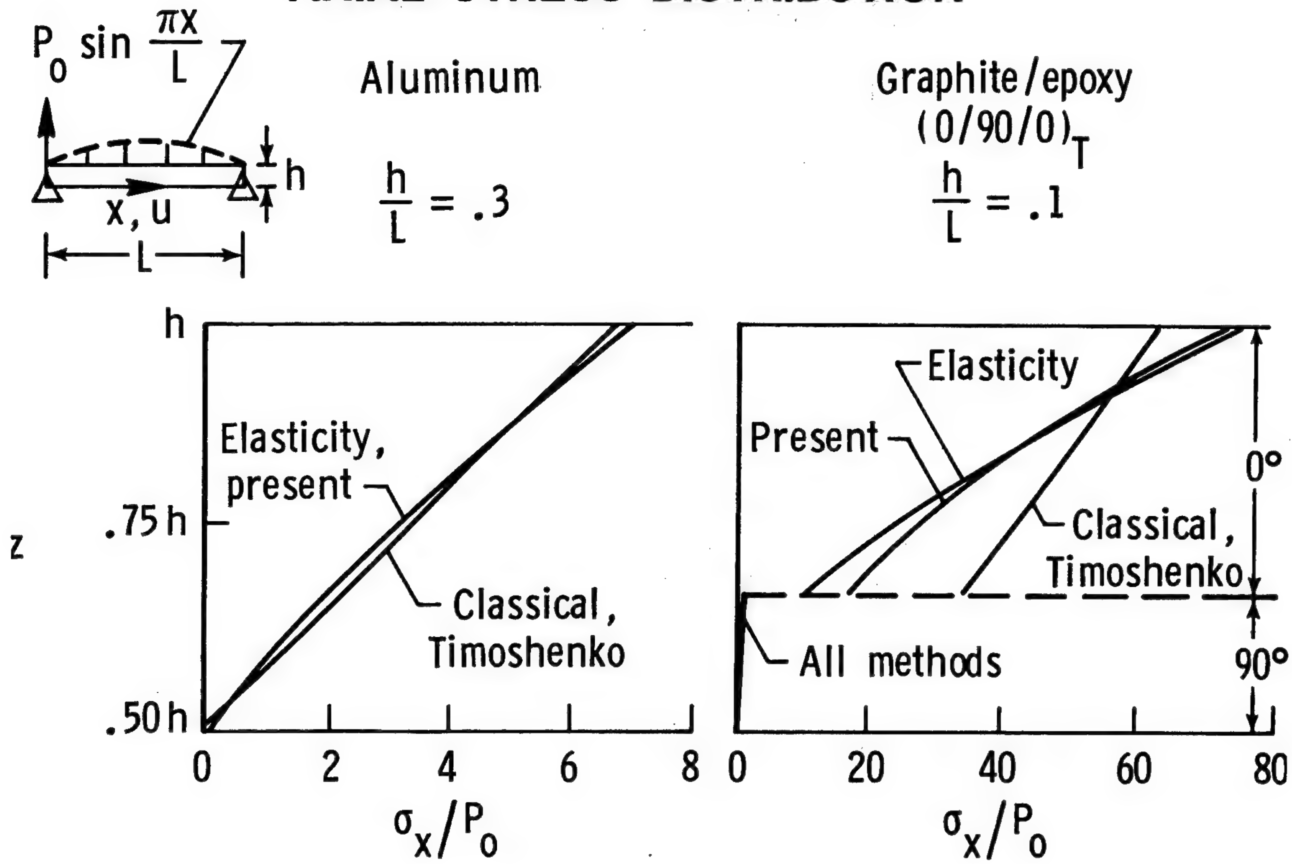
$$u(x,z) = u_a(x)\left(\frac{z}{h} - \frac{1}{2}\right) + u_1(x)\cos\frac{\pi z}{h}$$

$$w(x,z) = w_0(x) + w_1(x)\sin\frac{\pi z}{h}$$

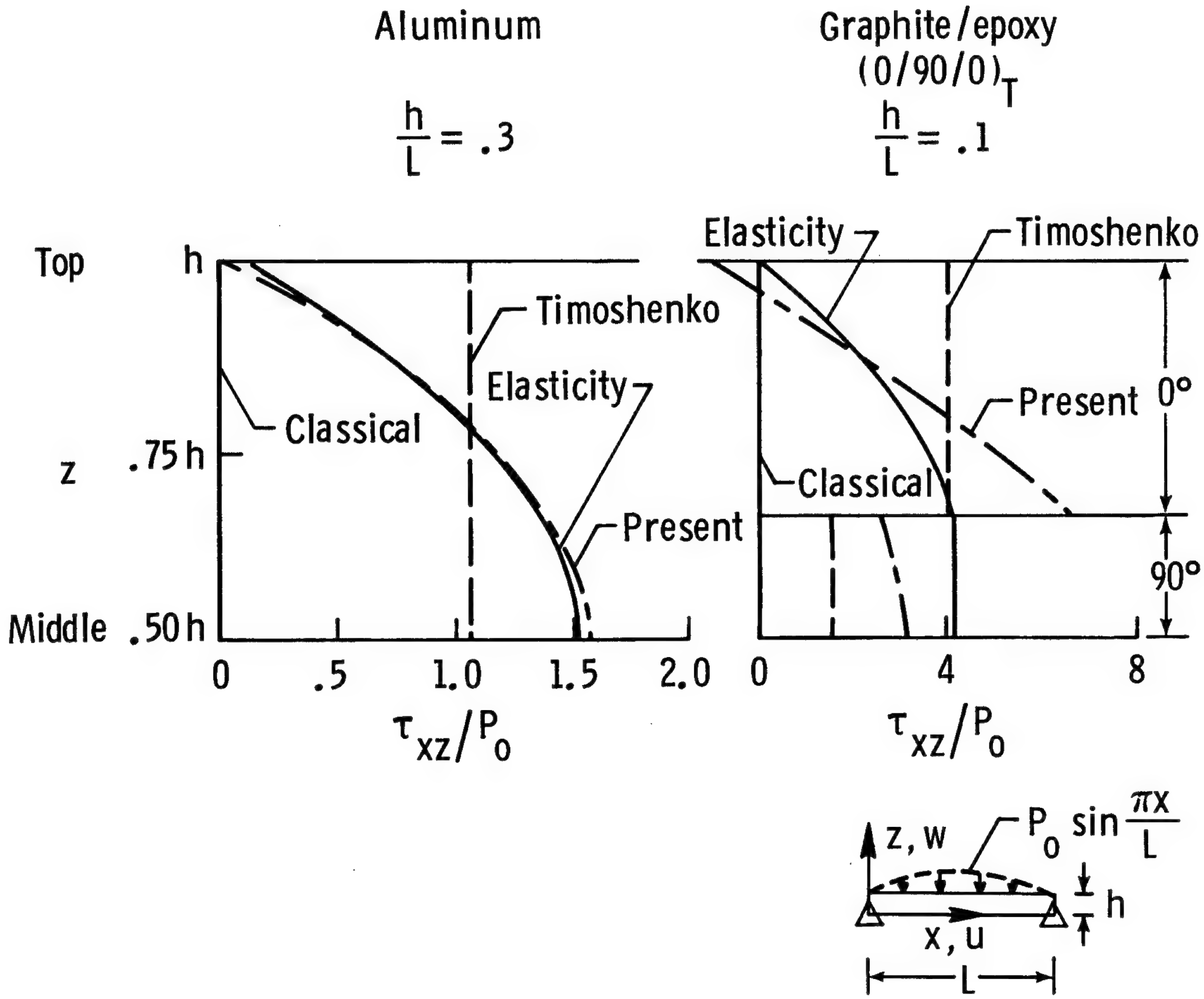
MAXIMUM DISPLACEMENT



AXIAL STRESS DISTRIBUTION

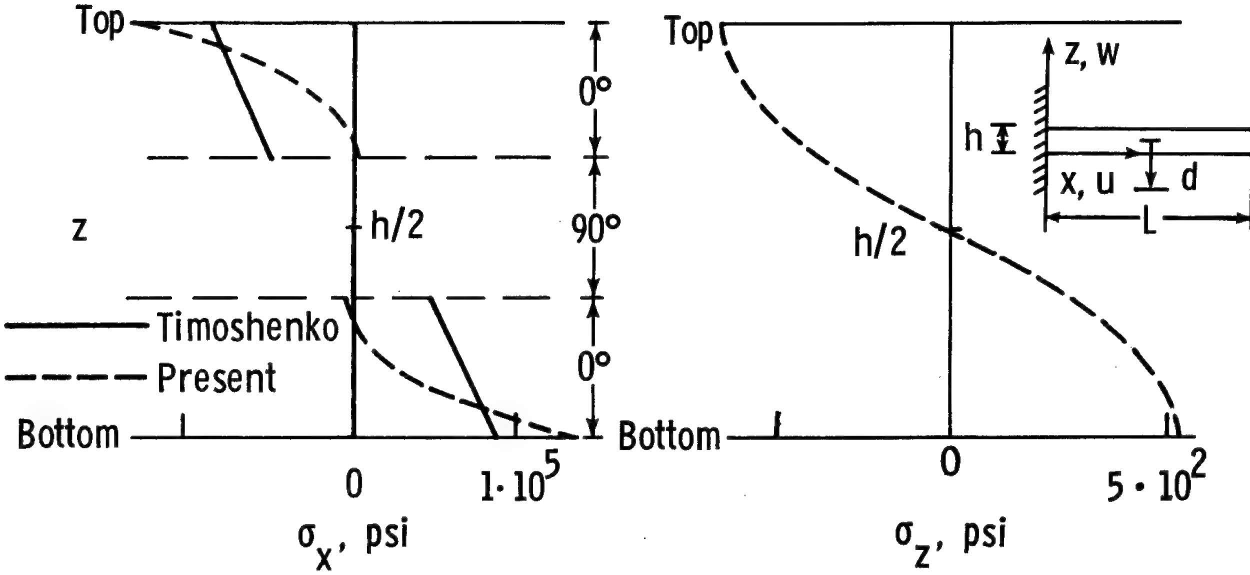


TRANSVERSE SHEAR STRESS DISTRIBUTION



MAXIMUM DIRECT STRESSES IN GRAPHITE/EPOXY BEAM $[0/90/0]_T$

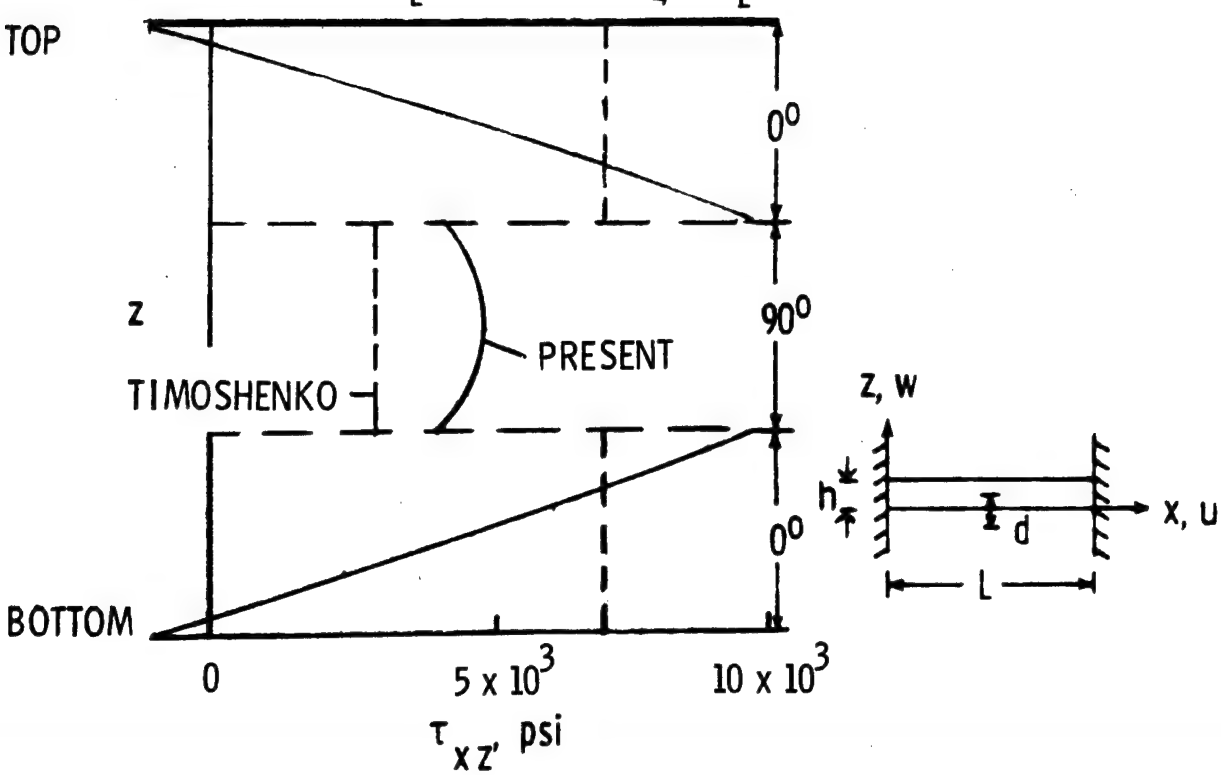
$\frac{h}{L} = 0.1 \quad \frac{d}{L} = 0.01$



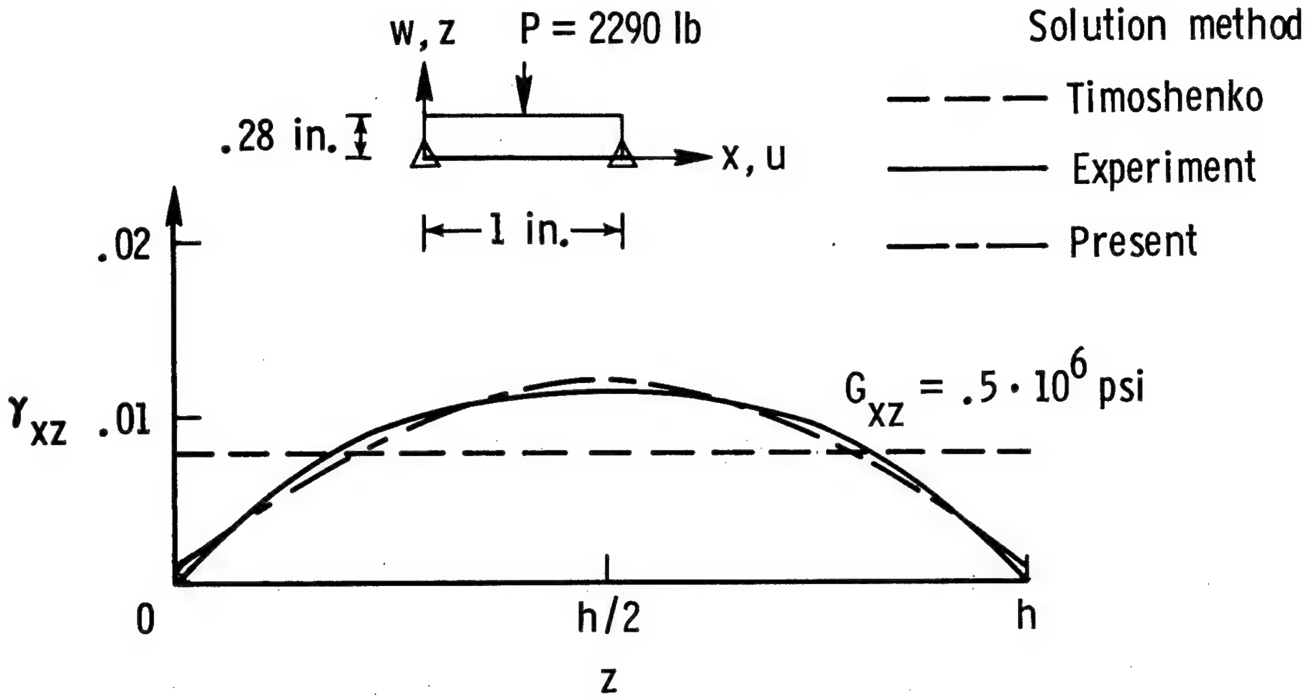
SHEAR STRESS IN GRAPHITE/EPOXY BEAM

$[0/90/0]_T$

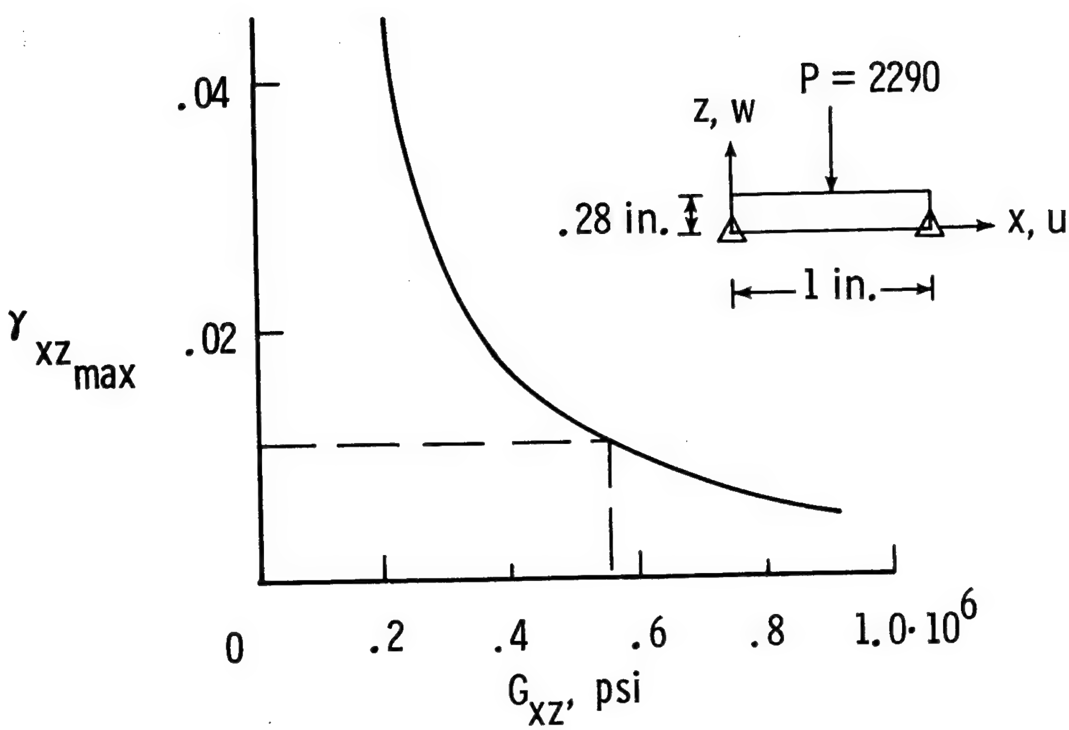
$\frac{h}{L} = 0.1 \quad x = \frac{L}{4} \quad \frac{d}{L} = 0.01$



EXPERIMENTAL AND ANALYTICAL COMPARISON OF UNIDIRECTIONAL T300/5208 GRAPHITE/EPOXY BEAM



VARYING G_{xz} TO FIND MAXIMUM SHEAR STRAIN WHICH FITS EXPERIMENT



CONCLUSIONS

- Present theory is accurate for isotropic and unidirectional beams
- Present theory when using potential energy is not adequate to predict transverse and shear stresses in beams which are not unidirectional
- Present work will be extended to complementary energy
- Present theory may prove useful for interpreting experimental data to find the through-the-thickness shear modulus

IMPROVEMENT AND OPTIMIZATION OF INTERNAL DAMPING IN FIBER REINFORCED COMPOSITE MATERIALS

R. F. Gibson and S. A. Suarez

University of Idaho
Moscow, Idaho 83843

C. T. Sun

University of Florida
Gainesville, Florida 32611

and

S. K. Chaturvedi

University of Missouri
Rolla, Missouri 65401

ABSTRACT

In a recent paper, it was concluded that the use of discontinuous fiber reinforcement appears to be a very attractive approach to the problem of improvement of internal damping in fiber reinforced polymer composite materials [1]. This conclusion was based primarily on a study of the different damping mechanisms operating in these materials and on the results of an earlier theoretical analysis of damping in aligned discontinuous fiber reinforced composites [2]. In Reference [2], it was shown that when the fiber damping is small, there is a theoretically optimum fiber aspect ratio where the loss modulus in a discontinuous aligned fiber composite is maximized. Since the predicted optimum aspect ratios were very low ($L/d < 20$, where L is fiber length and D is fiber diameter), it was expected that only greater-than-optimum values would be practically attainable.

The primary goals of the work discussed here were, (1) to fabricate and test composite specimens having fiber aspect ratios which were at least low enough that significant increases in damping could be measured, (2) to determine the effect of fiber orientation on damping (off-axis tests), (3) to compare measurements with predictions, and, (4) to determine how the analytical models could be modified to improve predictions. Graphite/epoxy, Kevlar/epoxy and boron/epoxy materials were tested in the fiber length studies, but only graphite/epoxy was tested off-axis. Preliminary results have been published in Reference [3].

Experimental results show that, as predicted by our analytical models, very low fiber aspect ratios are required to produce significant improvements in damping. Preliminary calculations also show that the marked improvement in observed damping of low fiber aspect ratio composites with increasing vibration frequency is predicted by the analysis. Curve-fitting of analytical models to experimental results show that the fiber stiffness is less than that specified by fiber manufacturers, that Kevlar fibers have a much higher loss factor than graphite or boron, and that the "effective" fiber aspect ratio for all composites tested is less than the length-to-diameter ratio of a single fiber. Within the range of fiber aspect ratios tested, the off-axis damping is essentially independent of aspect ratio. Measurements and predictions for graphite/epoxy show that as the off-axis angle θ increases from 0° to 90°, the loss factor increases, the stiffness decreases, but the loss modulus has a maximum in the range $10° \leq \theta \leq 15°$. Successful curve-fitting of off-axis experimental results required the use of anisotropic fiber properties in the micromechanics model. The off-axis behavior of Kevlar appears to be worthy of study, but difficulties with machining specimens are currently holding up such studies.

REFERENCES

1. Gibson, R. F., "Development of Damping Composite Materials," 1983 Advances in Aerospace Structures, Materials and Dynamics, AD-06, American Society of Mechanical Engineers (1983).

2. Gibson, R. F., Chaturvedi, S. K. and Sun, C. T., "Complex Moduli of Aligned Discontinuous Fibre-Reinforced Polymer Composites," Journal of Materials Science, 17, 3499-3509 (1982).

3. Gibson, R. F., Suarez, S. A., and Deobald, L. R., "Improvement of Damping in Fiber Reinforced Polymer Composites," Proceedings of Vibration Damping Workshop, sponsored by Air Force Wright Aeronautical Laboratories, Feb. 1984, Long Beach, California (in print).

IMPROVEMENT AND OPTIMIZATION OF

INTERNAL DAMPING IN FIBER REINFORCED

COMPOSITE MATERIALS

R. F. GIBSON AND S. A. SUAREZ

UNIVERSITY OF IDAHO

C. T. SUN

UNIVERSITY OF FLORIDA

S. K. CHATURVEDI

UNIVERSITY OF MISSOURI

SPONSORED BY AFOSR GRANTS 83-0154 AND 83-0156

PROGRAM MANAGER: D. R. ULRICH

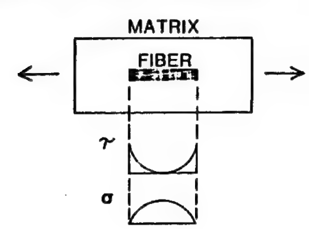
MOTIVATION:

1. PREVIOUS EXPERIMENTS SHOWED HIGH DAMPING IN SHORT FIBER COMPOSITES
2. DISCONTINUOUS CONSTRAINED VISCOELASTIC LAYER ANALOGY

Damping Mechanisms in Fiber Reinforced Polymers:

1. Viscoelastic behavior of matrix and/or fibers
2. Thermoelastic damping due to cyclic heat flow
3. Coulomb friction due to slip in unbonded regions of fiber/matrix interface
4. Dissipation caused by microscopic or macroscopic damage in composite

COX STRESS DISTRIBUTION ALONG DISCONTINUOUS FIBER

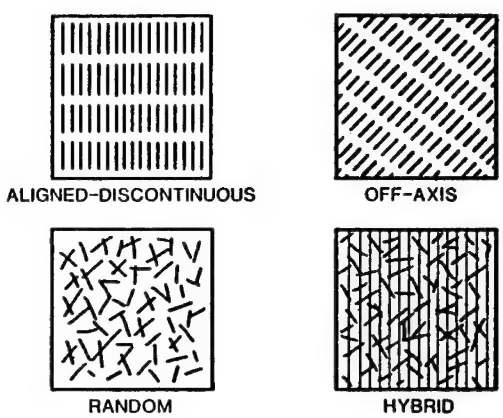


PARAMETERS CONSIDERED:

FIBER ASPECT RATIO, ℓ/d
FIBER VOLUME FRACTION, V_f
PACKING GEOMETRY
STIFFNESS RATIO, E_f/E_m
FIBER LOSS FACTOR, η_f
MATRIX LOSS FACTOR, η_m
FIBER ORIENTATION, θ

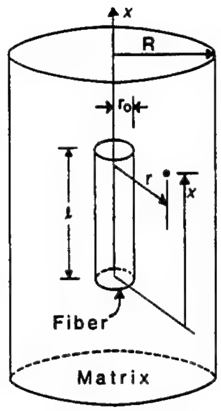
POLYMER COMPOSITES TO BE STUDIED

GRAPHITE / EPOXY
KEVLAR / EPOXY
WHISKER / EPOXY
MICROFIBER / EPOXY



ANALYSES:

1. ENERGY APPROACH
2. FORCE-BALANCE APPROACH



ELASTIC–VISCOELASTIC CORRESPONDENCE PRINCIPLE:

$$E_L^* = E_f^* \left[1 - \frac{\tanh \frac{\beta^* \ell}{2}}{\frac{\beta^* \ell}{2}} \right] v_f + E_m^* v_m$$

or

$$E_L' + i E_L'' = (E_f' + i E_f'') \left[1 - \frac{\tanh \frac{\beta^* \ell}{2}}{\frac{\beta^* \ell}{2}} \right] v_f$$

$$+ (E_m' + i E_m'') v_m$$

$$\text{where} \quad \tanh \frac{\beta^* \ell}{2} \simeq \tanh \frac{\beta \ell}{2} + \frac{i}{2} \frac{\beta \ell}{2} \frac{(\eta_{G_m} - \eta_f)}{\cosh^2 \frac{\beta \ell}{2}}$$

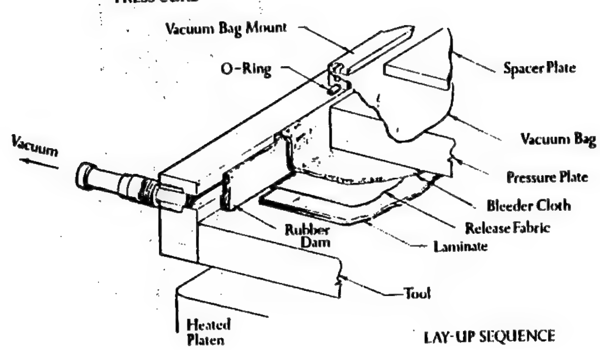
OFF-AXIS ANALYSIS

$$\frac{1}{E_x^*} = \frac{\cos^4\theta}{E_L^*} + \frac{\sin^4\theta}{E_T^*} + \left(\frac{1}{G_{LT}^*} - \frac{2\nu_{LT}^*}{E_L^*}\right) \sin^2\theta \cos^2\theta$$

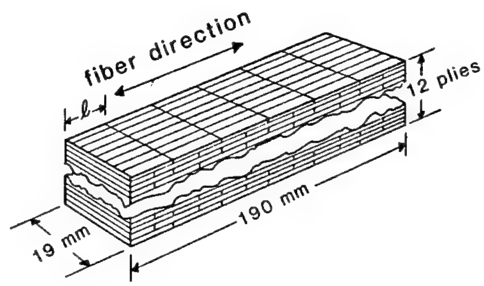
CORRESPONDENCE PRINCIPLE USED TO CONVERT TENSOR ELASTIC MODULUS TRANSFORMATION TO COMPLEX FORM.

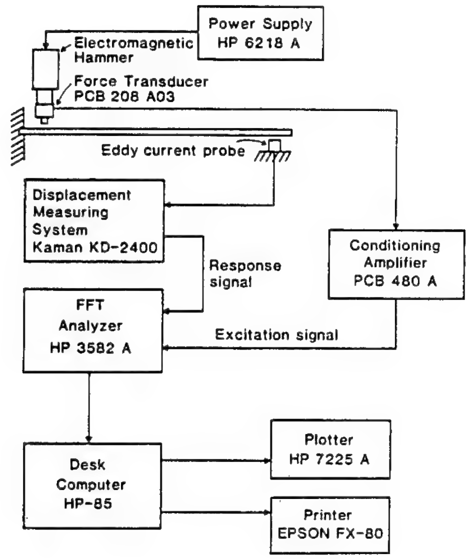
E_L^*, E_T^*, G_{LT}^*, ${}^{*}_{LT}$ FOUND FROM COMPLEX FORMS OF MICROMECHANICS EQUATIONS

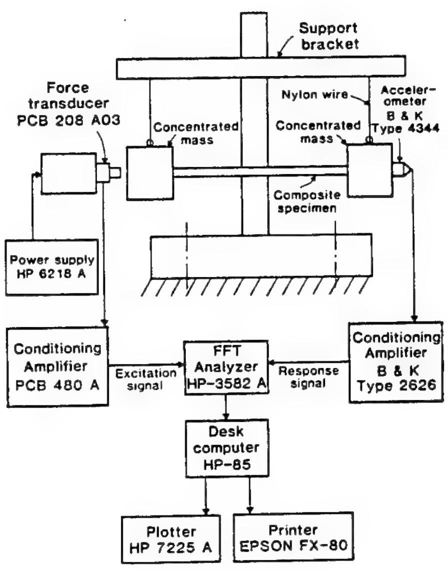
AUTOCLAVE-STYLE PRESS CURE

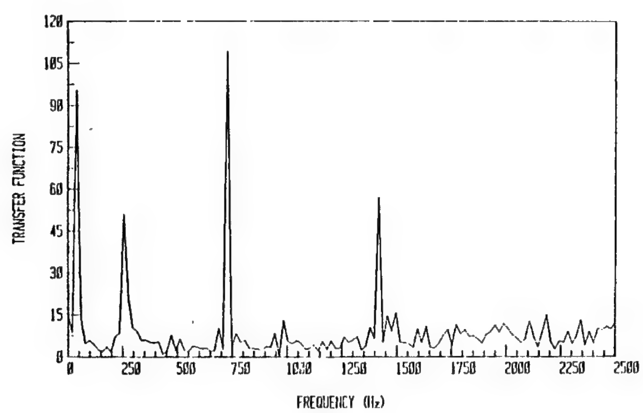


LAY-UP SEQUENCE


fiber direction
ℓ
12 plies
190 mm
19 mm


Power Supply
HP 6218 A
Electromagnetic
Hammer
Force Transducer
PCB 208 A03
Eddy current probe
Displacement
Measuring
System
Kaman KD-2400
Response
signal
Conditioning
Amplifier
PCB 480 A
FFT
Analyzer
HP 3582 A
Excitation signal
Plotter
HP 7225 A
Desk
Computer
HP-85
Printer
EPSON FX-80


Support
bracket
Force
transducer
PCB 208 A03
Nylon wire
Accelerometer
B & K
Type 4344
Concentrated
mass
Concentrated
mass
Composite
specimen
Power supply
HP 6218 A
Conditioning
Amplifier
PCB 480 A
Excitation
signal
FFT
Analyzer
HP-3582 A
Response
signal
Conditioning
Amplifier
B & K
Type 2626
Desk
computer
HP-85
Plotter
HP 7225 A
Printer
EPSON FX-80

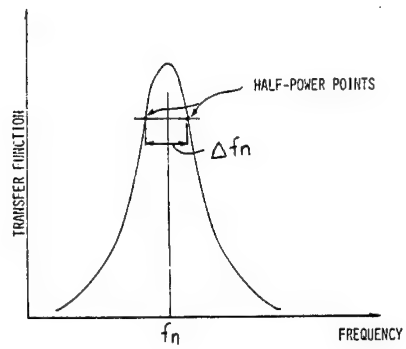

TRANSFER FUNCTION
120
105
90
75
60
45
30
15
0
0
250
500
750
1000
1250
1500
1750
2000
2250
2500
FREQUENCY (Hz)

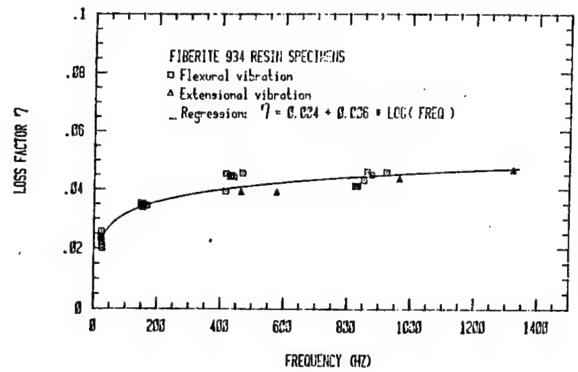
HALF-POWER POINTS

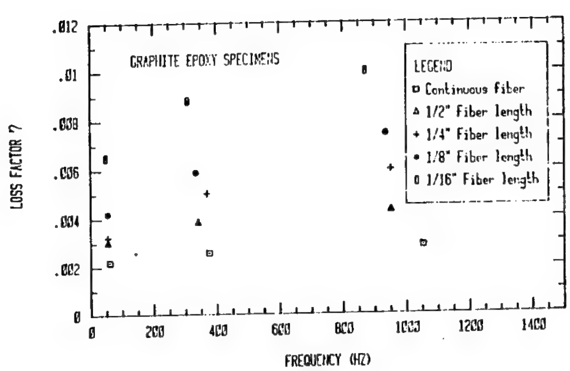
LOGARITHMIC SCALE: AT 3dB BELOW THE PEAK VALUE (AT RESONANCE FREQUENCY)

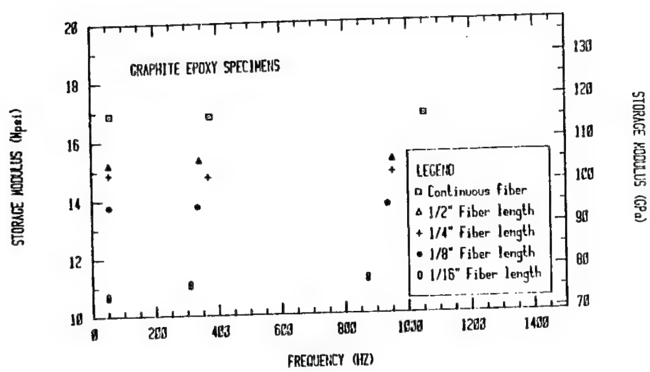
LINEAR SCALE: AT 0.707 THE PEAK VALUE

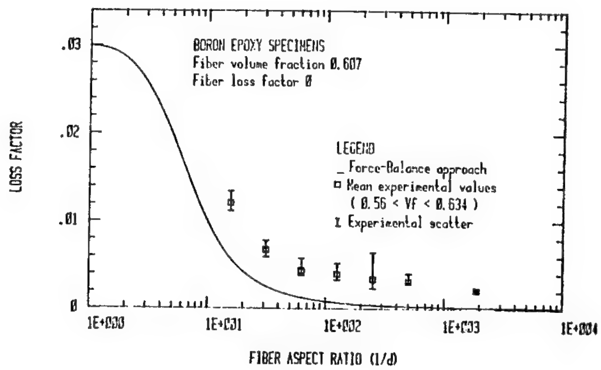
LOSS FACTOR = $\eta = \frac{\Delta F_N}{F_N}$

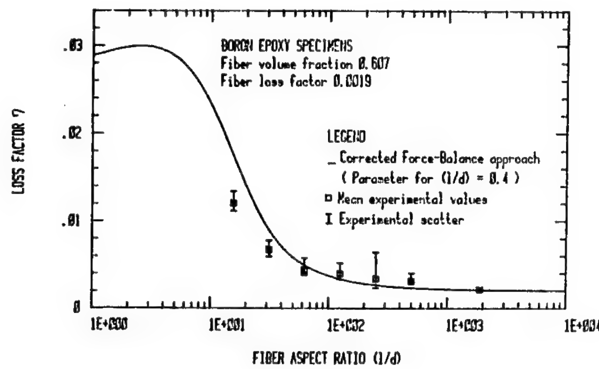


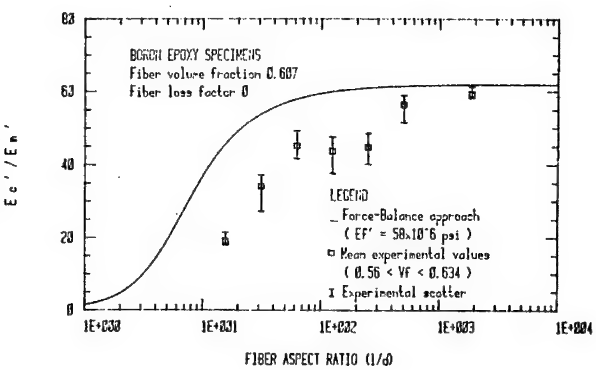


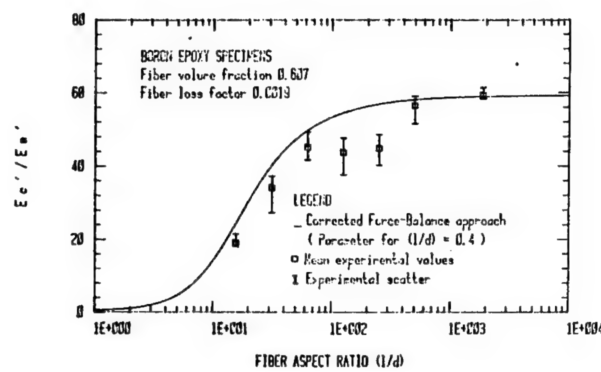


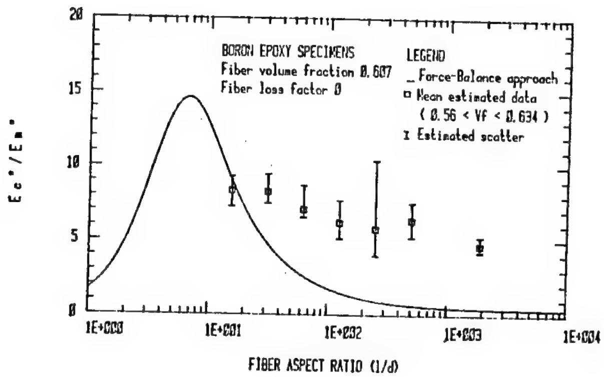


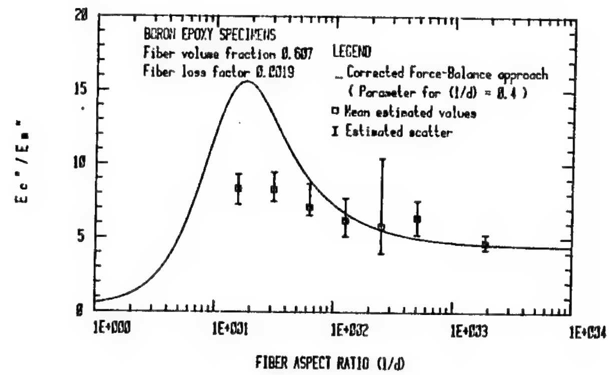

BORON EPOXY SPECIMENS
Fiber volume fraction 0.607
Fiber loss factor 0
LEGEND
_ Force-Balance approach
□ Mean experimental values
(0.56 < Vf < 0.634)
I Experimental scatter
LOSS FACTOR
.03
.02
.01
0
1E+000
1E+001
1E+002
1E+003
1E+004
FIBER ASPECT RATIO (l/d)

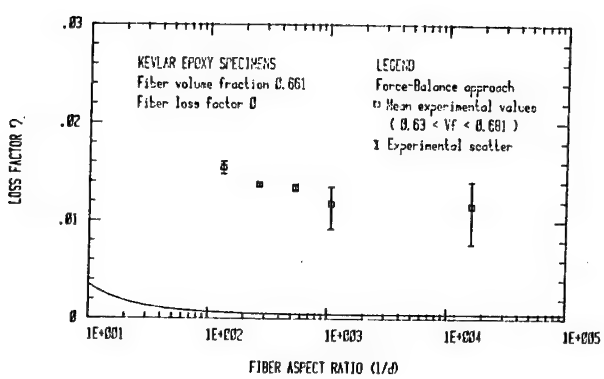

BORON EPOXY SPECIMENS
Fiber volume fraction 0.607
Fiber loss factor 0.0019
LEGEND
_ Corrected Force-Balance approach
(Parameter for (l/d) = 0.4)
□ Mean experimental values
I Experimental scatter
LOSS FACTOR
.03
.02
.01
0
1E+000
1E+001
1E+002
1E+003
1E+004
FIBER ASPECT RATIO (l/d)

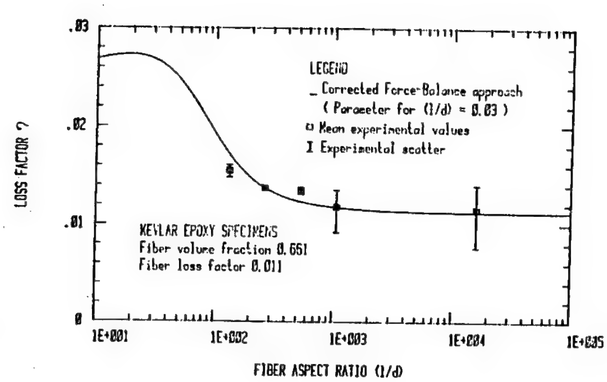

BORON EPOXY SPECIMENS
Fiber volume fraction 0.607
Fiber loss factor 0
LEGEND
_ Force-Balance approach
(EF' = 58x10^6 psi)
□ Mean experimental values
(0.56 < Vf < 0.634)
I Experimental scatter
Ec' / Em'
80
60
40
20
0
1E+000
1E+001
1E+002
1E+003
1E+004
FIBER ASPECT RATIO (l/d)

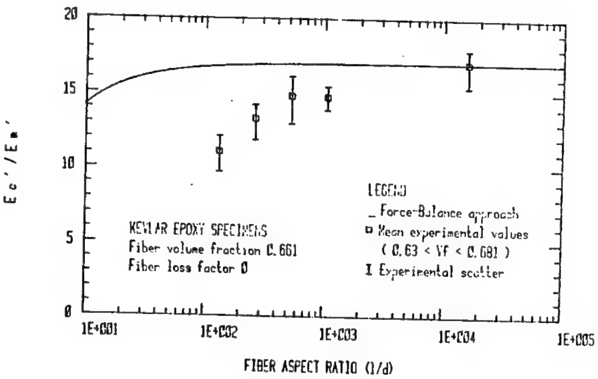

BORON EPOXY SPECIMENS
Fiber volume fraction 0.607
Fiber loss factor 0.0019
LEGEND
_ Corrected Force-Balance approach
(Parameter for (l/d) = 0.4)
□ Mean experimental values
I Experimental scatter
Ec' / Em'
80
60
40
20
0
1E+000
1E+001
1E+002
1E+003
1E+004
FIBER ASPECT RATIO (l/d)

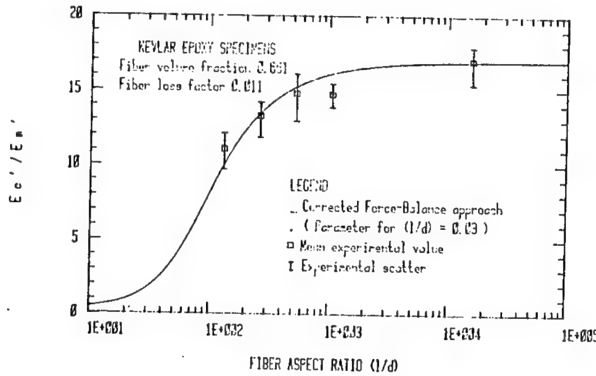

BORON EPOXY SPECIMENS
Fiber volume fraction 0.607
Fiber loss factor 0
LEGEND
_ Force-Balance approach
□ Mean estimated data
(0.56 < Vf < 0.634)
I Estimated scatter
FIBER ASPECT RATIO (l/d)

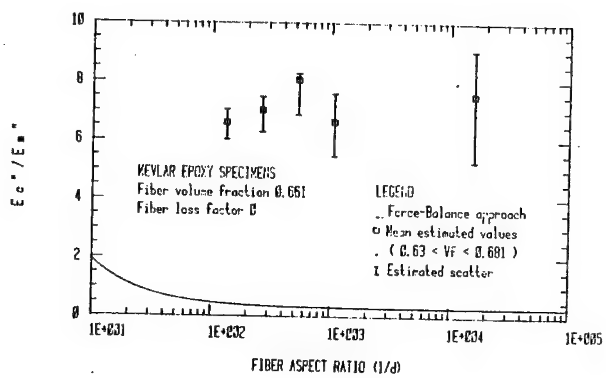

BORON EPOXY SPECIMENS
Fiber volume fraction 0.607
Fiber loss factor 0.0019
LEGEND
_ Corrected Force-Balance approach
(Parameter for (l/d) = 0.4)
□ Mean estimated values
I Estimated scatter
FIBER ASPECT RATIO (l/d)

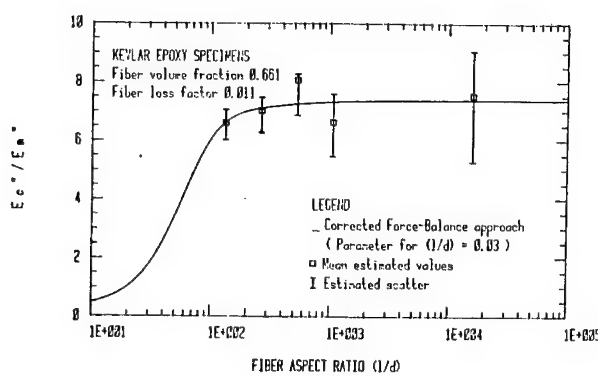

KEVLAR EPOXY SPECIMENS
Fiber volume fraction 0.661
Fiber loss factor 0
LEGEND
Force-Balance approach
□ Mean experimental values
(0.63 < Vf < 0.681)
I Experimental scatter
LOSS FACTOR ?
FIBER ASPECT RATIO (l/d)

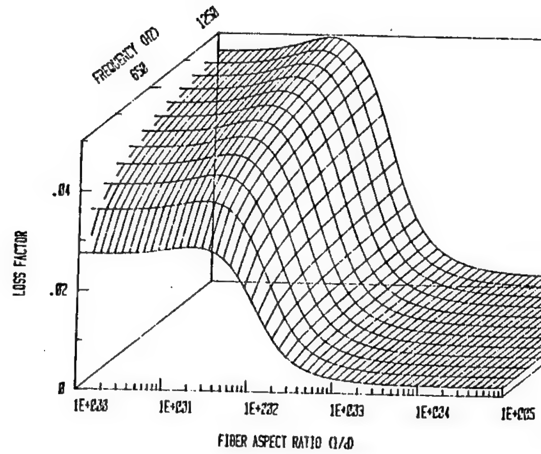

LEGEND
_ Corrected Force-Balance approach
(Parameter for (l/d) = 0.03)
□ Mean experimental values
I Experimental scatter
KEVLAR EPOXY SPECIMENS
Fiber volume fraction 0.661
Fiber loss factor 0.011
LOSS FACTOR ?
FIBER ASPECT RATIO (l/d)

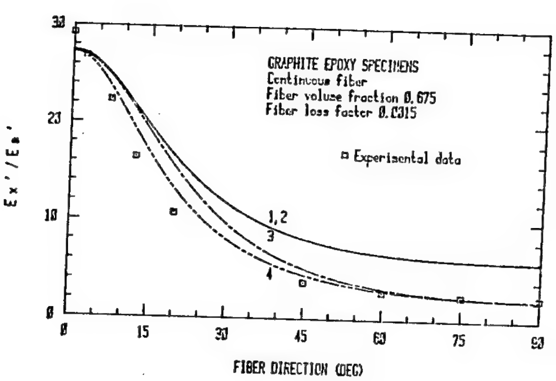

KEVLAR EPOXY SPECIMENS
Fiber volume fraction 0.661
Fiber loss factor 0
LEGEND
_ Force-Balance approach
□ Mean experimental values
(0.63 < Vf < 0.681)
I Experimental scatter
Ec'/Em'
FIBER ASPECT RATIO (l/d)

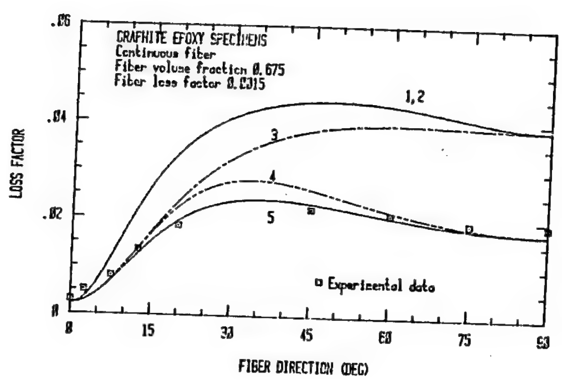

KEVLAR EPOXY SPECIMENS
Fiber volume fraction 0.661
Fiber loss factor 0.011
LEGEND
_ Corrected Force-Balance approach
(Parameter for (l/d) = 0.03)
□ Mean experimental value
I Experimental scatter
Ec'/Em'
FIBER ASPECT RATIO (l/d)


KEVLAR EPOXY SPECIMENS
Fiber volume fraction 0.661
Fiber loss factor 0
LEGEND
_ Force-Balance approach
□ Mean estimated values
(0.63 < Vf < 0.681)
I Estimated scatter
Ec"/Em"
FIBER ASPECT RATIO (l/d)


KEVLAR EPOXY SPECIMENS
Fiber volume fraction 0.661
Fiber loss factor 0.011
LEGEND
_ Corrected Force-Balance approach
(Parameter for (l/d) = 0.03)
□ Mean estimated values
I Estimated scatter
Ec"/Em"
FIBER ASPECT RATIO (l/d)


FREQUENCY (HZ)
650
1250
LOSS FACTOR
.04
.02
0
1E+000
1E+001
1E+002
1E+003
1E+004
1E+005
FIBER ASPECT RATIO (l/d)


GRAPHITE EPOXY SPECIMENS
Continuous fiber
Fiber volume fraction 0.675
Fiber loss factor 0.0015
Experimental data
1,2
3
4
Ex'/Em'
30
20
10
0
15
30
45
60
75
90
FIBER DIRECTION (DEG)


GRAPHITE EPOXY SPECIMENS
Continuous fiber
Fiber volume fraction 0.675
Fiber loss factor 0.0015
1,2
3
4
5
Experimental data
LOSS FACTOR
.06
.04
.02
0
15
30
45
60
75
90
FIBER DIRECTION (DEG)

ON INTERLAMINAR BEAM EXPERIMENTS FOR COMPOSITE MATERIALS

J. M. Whitney

Materials Laboratory
Air Force Wright Aeronautical Laboratories
Wright-Patterson Air Force Base, Ohio 45433

ABSTRACT

The short beam shear test has become a widely used method for characterizing the interlaminar failure resistance of fiber reinforced composites. This test method involves loading a beam under 3-point bending with the dimensions such that an interlaminar shear failure is induced. The simplicity of the test method makes it a very popular materials screening tool. As pointed out in the title of ASTM Standard D-2344, this method measures the "apparent" interlaminar shear strength of composite materials. Thus, the short beam shear method is not appropriate for generating design information. Despite such warnings, data generated from this method have had a tendency to be used as design allowables.

A second limitation on this test method, in conjunction with graphite-epoxy materials, raises serious doubts about its usefulness even as a materials screening tool. In particular, when used in conjunction with thin unidirectional beams, which is common practice, the test method does not yield interlaminar failures. Such data is often reported in the literature without mentioning the failure mode, leaving the reader to believe that the desired interlaminar failure was attained. Alternatives to the classical short beam shear method have been proposed [1]. These involve thick short beam shear specimens and beam specimens under 4-point bending at quarter points with dimensions chosen to induce an interlaminar failure mode. Although such alternatives do yield interlaminar failures, the mechanism which induces these failures needs to be thoroughly examined.

In the present paper experimental data is combined with a stress analysis and photomicrographs of actual failure modes to bring interlaminar beam test methods into a clearer perspective.

An analytical solution utilizing classical theory of elasticity is available [2] for performing a detailed stress analysis of the various interlaminar beam specimens currently in use. The solution method utilizes a stress function in the form

$$\phi = c_1 y^2 + c_2 x^2 + c_3 x^3 + \sum_{m=1}^{\infty} f_m(y) \cos \frac{2m\pi x}{a}$$

where a is the length of the beam, including overhang. The load and support conditions are represented by a Fourier series. The resulting concentrated loads are modeled as a uniform stress distributed over a very small area. Forces and moments at the ends of the beam due to normal bending stresses vanish. For overhang lengths representative of laboratory specimens, however, the bending stresses actually vanish. Thus, from a practical standpoint, the solution satisfies all of the required boundary conditions.

Experimental data is presented in conjunction with unidirectional graphite-epoxy and graphite-thermoplastic composites. The thermoplastic matrix is polyetheretherketone (PEEK), which is a ductile resin. Failure loads and modes of failure will be shown to be a function of specimen geometry and load, e.g., 3-point load and 4-point load. The stress analysis indicates an absolute specimen size effect.

REFERENCES

1. C. E. Browning, F. L. Abrams, and J. M. Whitney, "A Four-Point Shear Test for Graphite/Epoxy Composites," Composite Materials: Quality Assurance, and Processing, ASTM STP 797, C. E. Browning, Editor, American Society for Testing and Materials, 1983, pp. 54-74.

2. J. M. Whitney, "Elasticity Analysis of Orthotropic Beams Under Concentrated Loads," Composite Science and Technology, September 1984.

ON INTERLAMINAR BEAM EXPERIMENTS FOR COMPOSITE MATERIALS

J. M. WHITNEY
C. E. BROWNING

MATERIALS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES

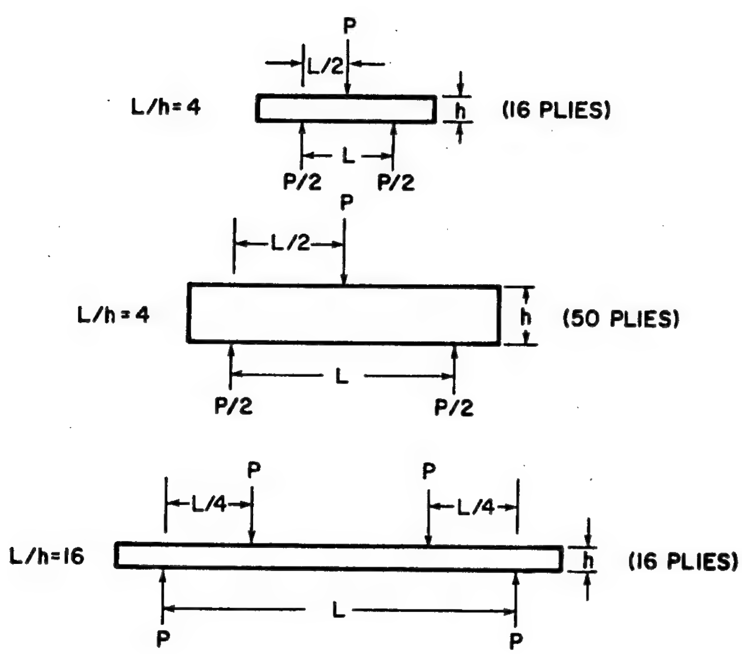
OBJECTIVE

TO ASSESS THE VALIDITY OF BEAM BENDING TESTS FOR CHARACTERIZING INTERLAMINAR SHEAR STRENGTH OF UNIDIRECTIONAL COMPOSITE MATERIALS

APPROACH

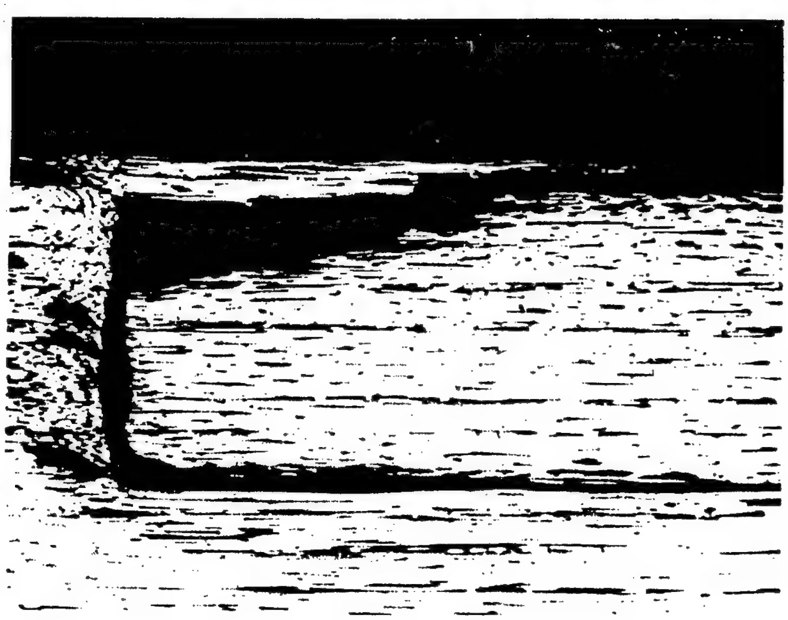
- OBTAIN DATA ON 3-POINT AND 4-POINT BEND SPECIMENS OF VARYING GEOMETRY
- OBTAIN PHOTOMICROGRAPHS IN CONJUNCTION WITH FAILED SPECIMENS IN ORDER TO OBSERVE ACTUAL FAILURE MODES
- PERFORM STRESS ANALYSIS USING CLASSICAL THEORY OF ELASTICITY

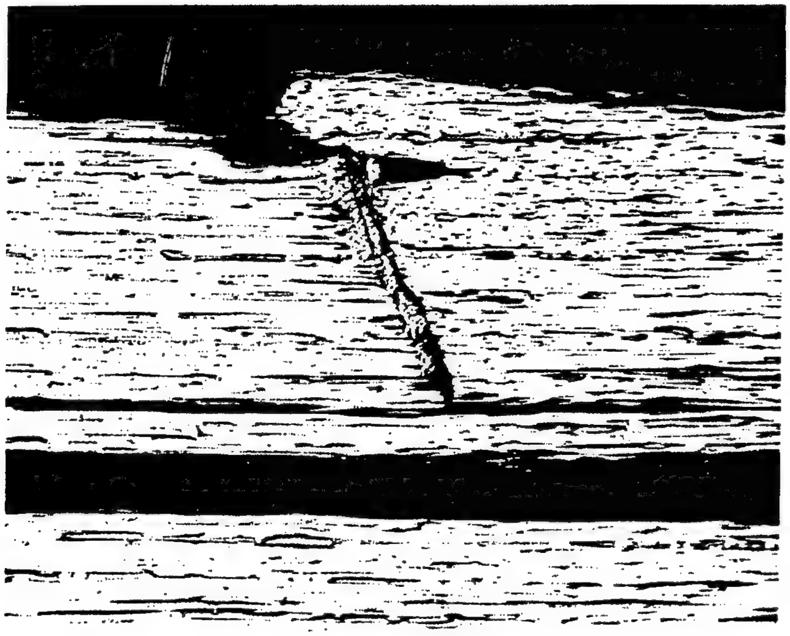
INTERLAMINAR BEAM TESTS

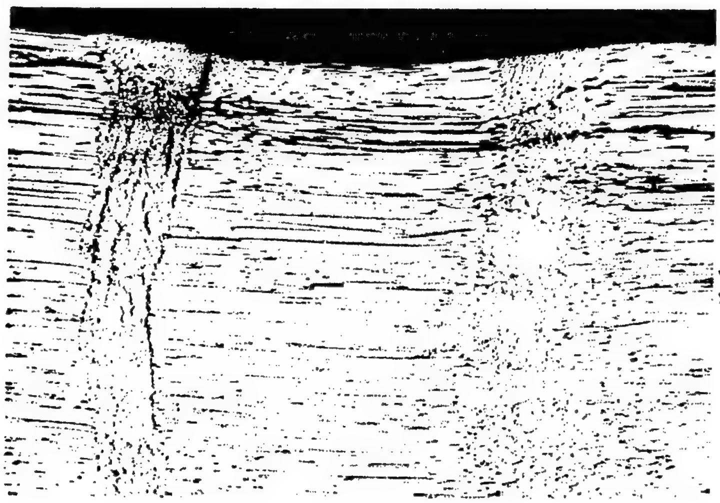


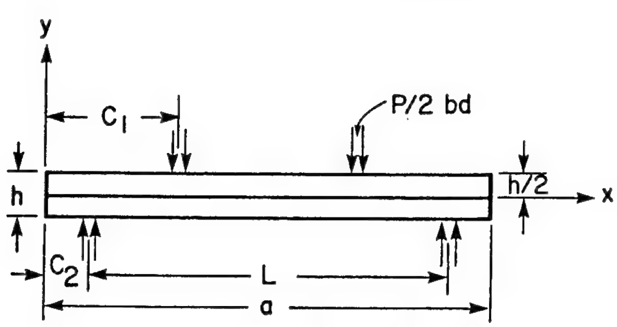
EXPERIMENTAL DATA

TEST	MATERIAL	PLIES	REPL.	L/h	τ, MPa (KSI)	MODE
SBS	AS-1/3502	16	24	4	102 (14.8)	C
SBS	AS-1/3502	50	22	4	96 (13.9)	I
FPS	AS-1/3502	16	51	16	88 (12.7)	I
FPS	AS-1/3502	24	25	16	81 (11.7)	I
FPS	X-AS/PEEK	34	2	8	115 (16.6)	T, C









$$\sigma_y(x,\ h/2) = q_1(x), \quad \sigma_y(x,\ -h/2) = -q_2(x)$$

$$\tau_{xy}(x,\ \pm h/2) = 0$$

$$\tau_{xy}(0,\ y) = \tau_{xy}(a,\ y) = 0$$

$$\int_{-h/2}^{h/2} \sigma_x(0,\ y)\, dy = \int_{-h/2}^{h/2} \sigma_x(0,\ y)\, y\, dy = 0$$

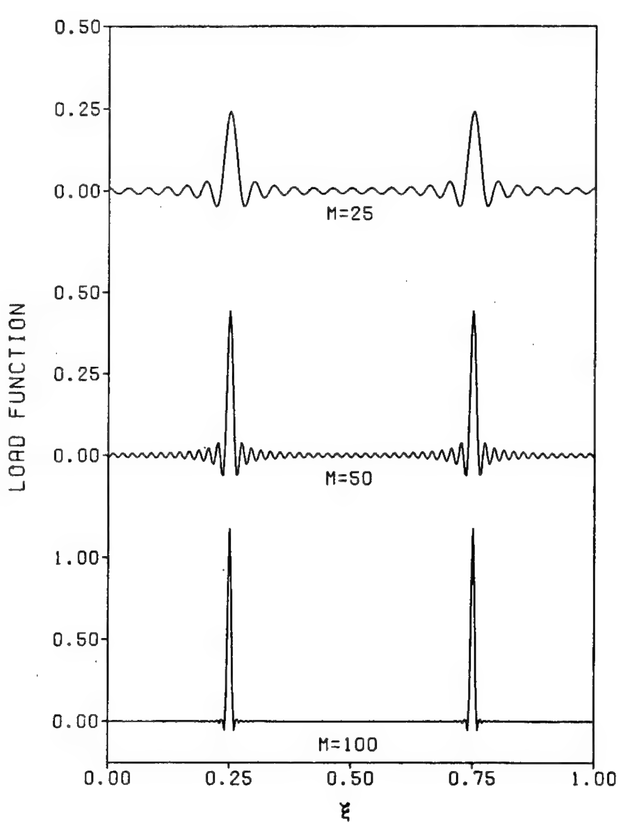
ELASTICITY SOLUTION

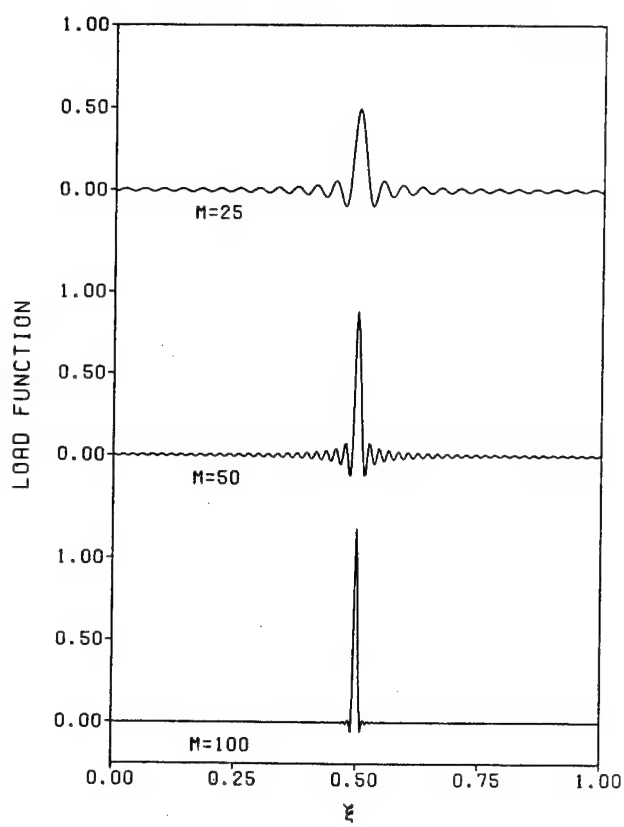
LOADS REPRESENTED BY FOURIER COSINE SERIES

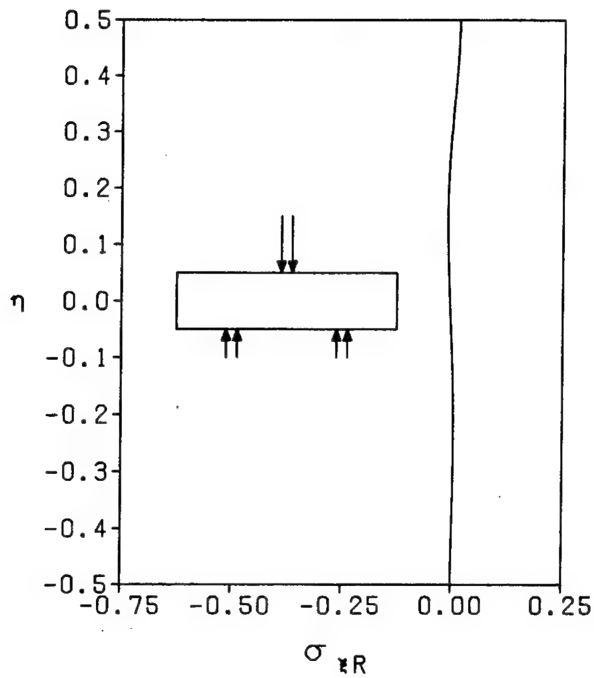
$$\sigma_x = \frac{\partial^2 \phi}{\partial y^2}, \quad \sigma_y = \frac{\partial^2 \phi}{\partial x^2}, \quad \tau_{xy} = -\frac{\partial^2 \phi}{\partial x \partial y}$$

$$\phi = \frac{A_0}{2} y^2 + \frac{B_0}{6} y^3 + \frac{C_0}{2} x^2 + \sum_{m=1}^{M} \sum_{i=1}^{2} \left[A_{mi} \frac{\text{COSH}(2\lambda_i\, m\pi y/a)}{\text{COSH}(\lambda_i\, m\pi h/a)} + B_{mi} \frac{\text{SINH}(2\lambda_i\, m\pi y/a)}{\text{SINH}(\lambda_i\, m\pi h/a)} \right] \text{COS}(2\, m\pi x/a)$$

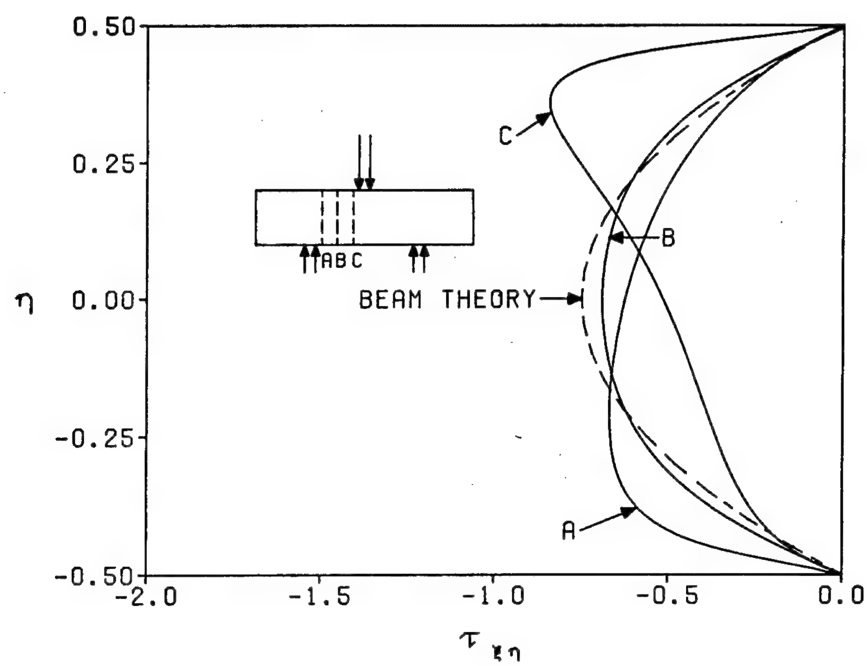
λ_i = ROOTS OF CHARACTERISTIC EQUATION INVOLVING ELASTIC CONSTANTS



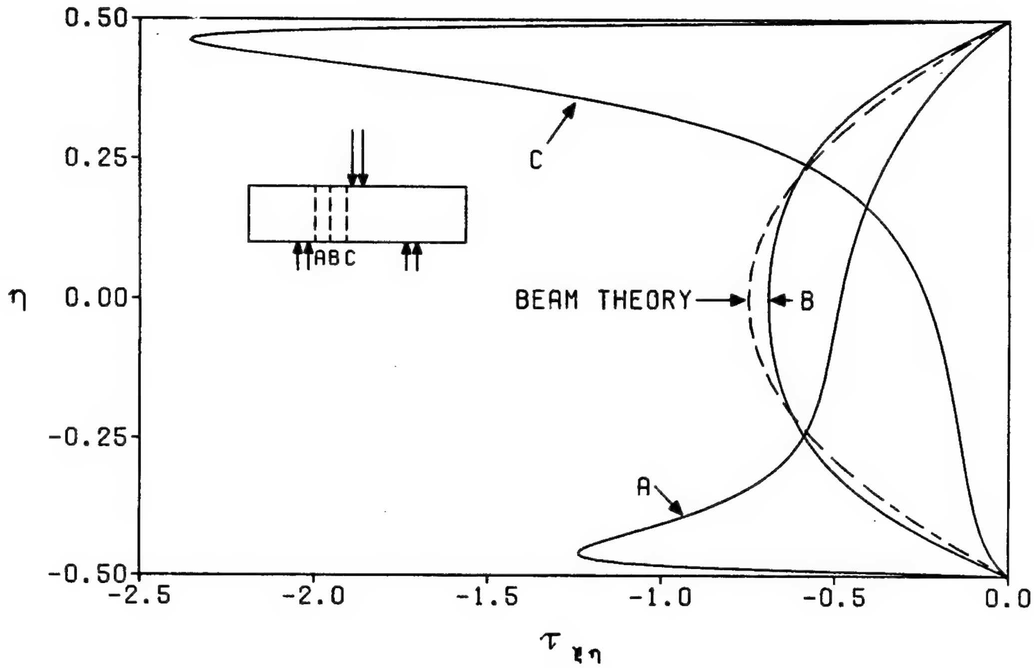


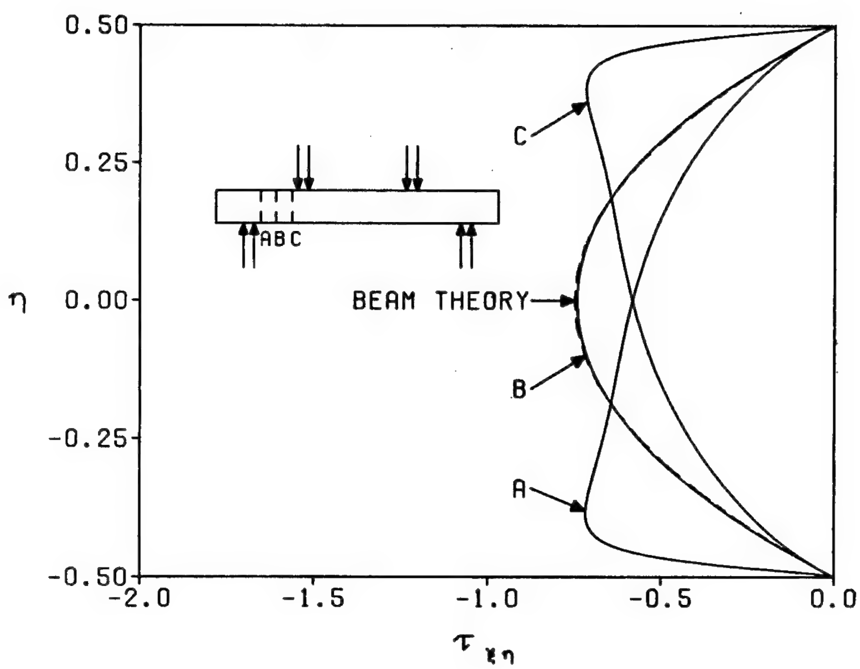


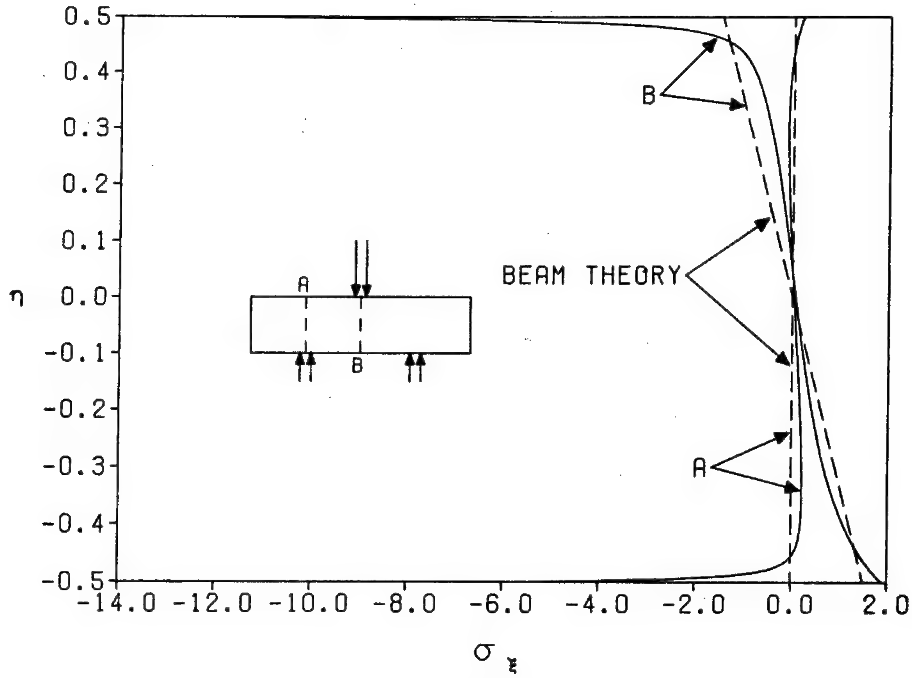
0.5
0.4
0.3
0.2
0.1
η
0.0
-0.1
-0.2
-0.3
-0.4
-0.5
-0.75
-0.50
-0.25
0.00
0.25
σ ξR

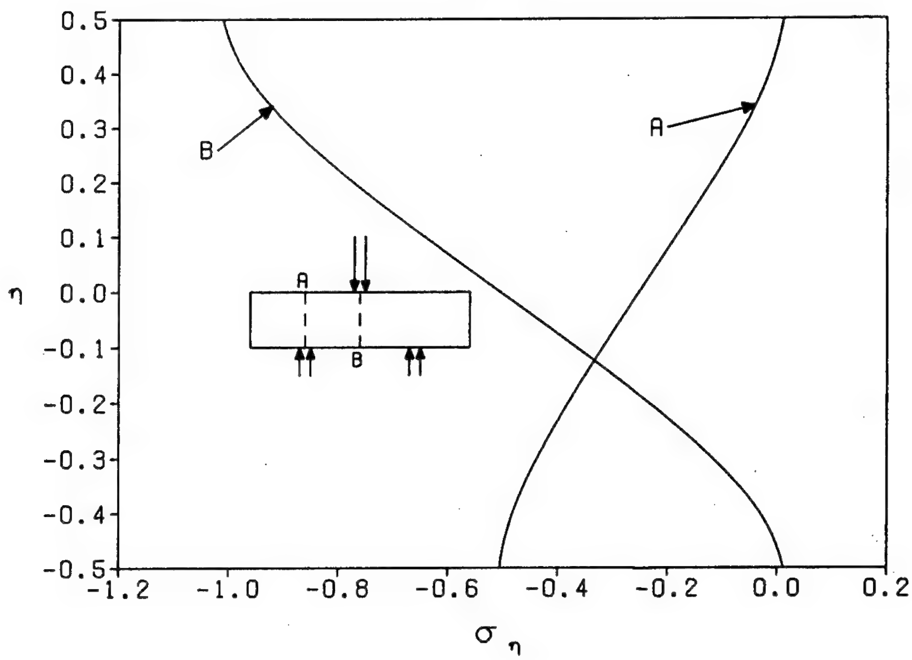


0.50
0.25
η
0.00
-0.25
-0.50
-2.0
-1.5
-1.0
-0.5
0.0
τ ξη
C
B
A
ABC
BEAM THEORY


0.50
0.25
0.00
-0.25
-0.50
-2.5
-2.0
-1.5
-1.0
-0.5
0.0
C
A B C
BEAM THEORY
B
A


0.50
0.25
0.00
-0.25
-0.50
-2.0
-1.5
-1.0
-0.5
0.0
C
A B C
BEAM THEORY
B
A





CONCLUSIONS

INTERLAMINAR BEAM TESTS

- STATE OF STRESS VERY COMPLEX AND NOT PREDICTABLE BY CLASSICAL BEAM THEORY
- PHOTOMICROGRAPHS INDICATE FAILURE MODE IS NOT AS ASSUMED
- TEST METHOD INADEQUATE FOR PROVIDING DESIRED INFORMATION RELATIVE TO INTERLAMINAR FAILURE RESISTANCE

DAMAGE MECHANISMS AND MOISTURE EFFECTS IN COMPOSITES AND POLYMERS*

Y. Weitsman

Mechanics & Materials Center
Civil Engineering Department
Texas A&M University

ABSTRACT

This presentation summarizes several investigations that were performed during the past two years, divided into two major topics.

The first topic concerned the modelling and analysis of crazes, since such damage mechanisms are likely to develop in resins that are considered for aerospace applications.

Crazes are thin, elongnated defects of length to thickness ratio of $0(10^3)$ that develop in many polymers. They consist of interconnected voids that are transversed by thin fibrils, which span the gap between opposite faces of the bulk polymer. In the analyses [1]-[4], the fibrillated craze region was modelled as a uni-dimensional material of given force-displacement relation. This relation was considered as linearly elastic or viscoelastic. In addition, a non-linear behavior with hysteresis upon unloading as observed experimentally, was also considered. The surrounding bulk polymer was considered to respond linearly, either elastically or viscoelastically.

The formulations for a single, dominant craze - with or without an internal crack - in an extended medium of bulk polymeric material employed singular integral equations akin to fracture mechanics. In all circumstances a square-root stress singularity developed at the craze tips, but the stress intensity factors were less than 25% of those associated with cracks of equal lengths. The abovementioned non-physical singularity is removable by means of Barenblatt-type tip zones with stresses of finite magnitudes. Such tip zones, of relatively short lengths and poorly defined structure are in fact known to exist in real circumstances.

The evaluation of stress intensity factors at the tips of crazes enables the establishment of criteria for their growth and arrest. For instance, it was found that the stop-start behavior observed in craze growth can be explained by means of the viscoelastic responses of both craze fibrils and bulk polymer. Furthermore, it was noted that residual stresses develop near the tips of crazes due to the hysteresis-loop under unloading. These stresses can be related to behavior under cyclic loads.

The second major topic of research concerned the response of fiber-reinforced composites under moisture, with special emphasis on damage formation under cycling humidities [5]-[8].

Two investigations [5], [7] involved square lay-ups of $[0°/90°/0°_4/90°_4/0°/90°]_T$ AS4/3502 gr/ep composites, which were exposed to various levels of relative humidity and temperature. Weight gains were monitored and compared against computations based on Fick's law, and curvatures were compared against predictions of linear elasticity and viscoelasticity. It was noted that during the first absorption cycle Fick's law and viscoelasticity theory approximated data to a satisfactory degree. However, significant departures between theory and experiment were noted during moisture desorption or when the specimens were exposed to fluctuating relative humidities. Detailed inspection revealed that during desorption damage develops in the form of debonding between matrix and fibers and grows by coalescence of initially disjoint and localized debonds.

*Investigations conducted under Contract N00014-82-K-0562 from the Mechanics Division, Engineering Sciences Directorate of ONR and under Contract F49620-82-C-0057 from the Office of Aerospace Research of AFOSR.

Another investigation [6] was performed to detect effects of stress on moisture absorption. The investigation involved uni-directional coupons of F155 Hexcel gr/ep composites loaded transversely at various stress levels and exposed to ambient relative humidities. An obvious effect of stress was noted, but further validation is required due to excessive scatter in data.

The last investigation [8] concerned the moisture diffusion in hybrids. Specimens were formed of uni-directional F155 and F185 gr/ep composites that can be co-cured simultaneously and exposed to ambient humidity. An analytic and computational method was developed, based on the concept of the "extended inner product", and solutions were obtained for Fickean diffusion. Comparisons between data and theory were uncertain, probably because the diffusion process in Hexcel composites is non-Fickean.

REFERENCES

1. Y. Weitsman, "Viscoelastic Effects on Stresses and Stress Intensities in Crazes," in Recent Developments in Applied Mathematics, F.F. Ling, and I.G. Tadjbakhsh, Editors, Rensselaer Press, 1983, pp. 220-234.

2. J.R. Walton and Y. Weitsman, "Deformation and Stress Intensities Due to a Craze in an Extended Elastic Material." Journal of Applied Mechanics, ASME, Vol. 51, No. 1, pp. 84-92, 1984.

3. Y. Weitsman, "Non-Linear Analysis of Crazes," Report MM4762-84-11, Texas A&M University, 1984.

4. J.R. Walton, "The Analysis of an Edge Craze," in preparation.

5. B.D. Harper and Y. Weitsman, "On the Effects of Environmental Conditioning on Residual Stresses in Composite Laminates," Report MM4665-84-10, Texas A&M University, April 1984. To appear in the Int. J. of Solids and Structures.

6. E.J. Porth, "Effect of an External Stress on Moisture Diffusion in Composite Materials," Report MM4665-83-15, Texas A&M University, December 1983.

7. S.P. Jackson, "Hygrothermal Effects in an Anti-Symmetric Cross-Ply Graphite/Epoxy Material," Report MM4665-84-7, Texas A&M University, May 1984.

8. D.L. Clark, "Moisture Absorption in Hybrid Composites," Report MM4665-83-16, Texas A&M University, December 1983.

CONCLUSIONS

1. Growth and arrest mechanisms of crazes can be predicted by means of mechanics models akin to fracture mechanics.

2. Damage due to environmental effects occurs by weakening and debonding of the fiber/matrix interfaces.

3. The most severe damage occurs when composites are exposed to fluctuating relative humidities, with the drying stage more critical than the absorption stage.

4. Damage due to moisture depends on the history of environmental exposure and not simply on the largest content of moisture weight-gain.

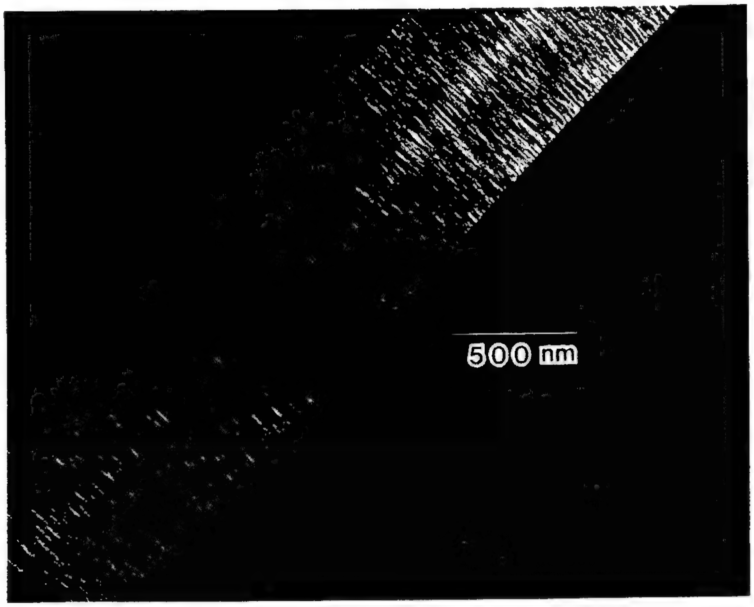


Figure 1. The central portion of a "mature case".

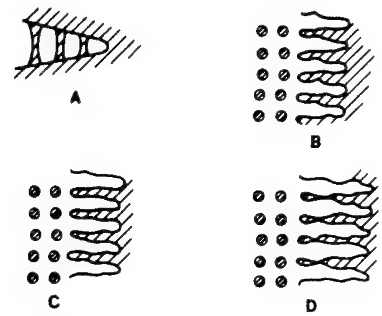


Figure 2. A schematic drawing of the craze tip and its growth process.

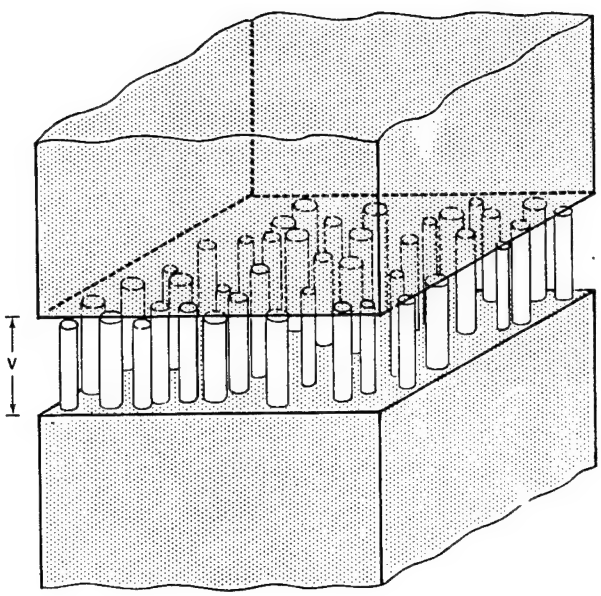


Figure 3. A craze model showing parallel fibrils connecting opposite faces of the bulk polymer.

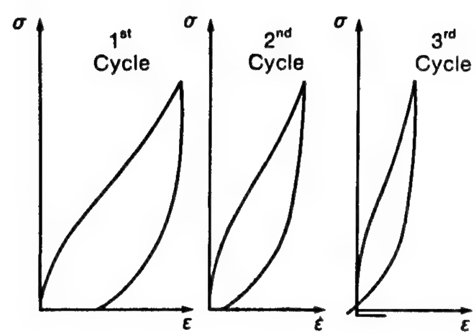


Figure 4. A sketch of a typical stress-strain response of the craze material.

Governing Equations for Craze Model

$$\tau_{XY}(X,0) = 0 \qquad -\infty < X < \infty$$

$$V(X,0) = 0 \qquad |X| > a$$

$$\sigma_Y(X,0) = -T + F[V(X,0),X] \qquad |X| < a \tag{1}$$

$$\sigma_Y(X,Y) \to 0 \qquad |Y| \to \infty$$

$$\sigma_Y(X,0) = \frac{G}{1-\nu}\,\frac{1}{\pi}\int_{-a}^{a} \frac{\partial V(\xi,0)}{\partial \xi}\,\frac{d\xi}{\xi - X} \tag{2}$$

$$T = F(V,X) - \frac{G}{1-\nu}\,\frac{1}{\pi}\int_{-a}^{a} \frac{\partial V}{\partial \xi}\,\frac{d\xi}{\xi - X} \tag{3}$$

$$T = F(V,X) - \frac{G}{1-\nu}\,\frac{1}{\pi}\,\frac{d}{dX}\int_{-a}^{a} \frac{V}{\xi - X}\,d\xi \tag{4}$$

where $\sigma = F(V)$ is stress-displacement relation for craze X,Y cartesian coordinates σ_Y, τ_{XY} normal and shear stresses, T remote load as $|Y| \to \infty$, V displacement in Y direction, a half craze length along X axis, and G, ν are shear modulus and Poisson's Ratio of the bulk material.

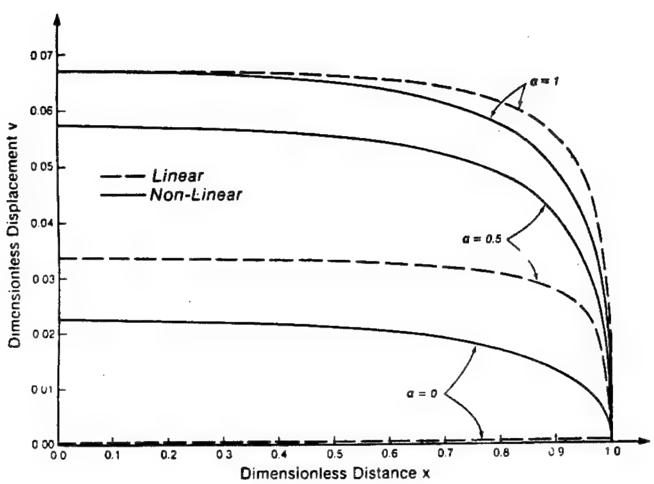


Figure 5. Non-dimensional opening displacement v(x) vs. the non-dimensional distance x during first unloading cycle. Cases of full load (α=1), partial unloading (α=0.5) and complete unloading (α=0). Non-linear results (solid lines) and linear values (dashed lines).

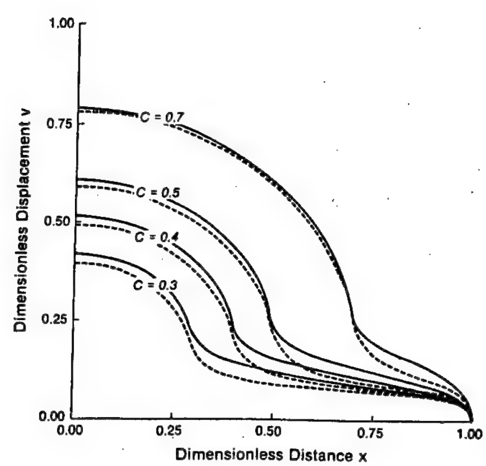


Figure 6. Compartive opening displacement profiles v(x) vs. x for selected values of c. Non-linear results (solid lines) and linear values (dashed lines).

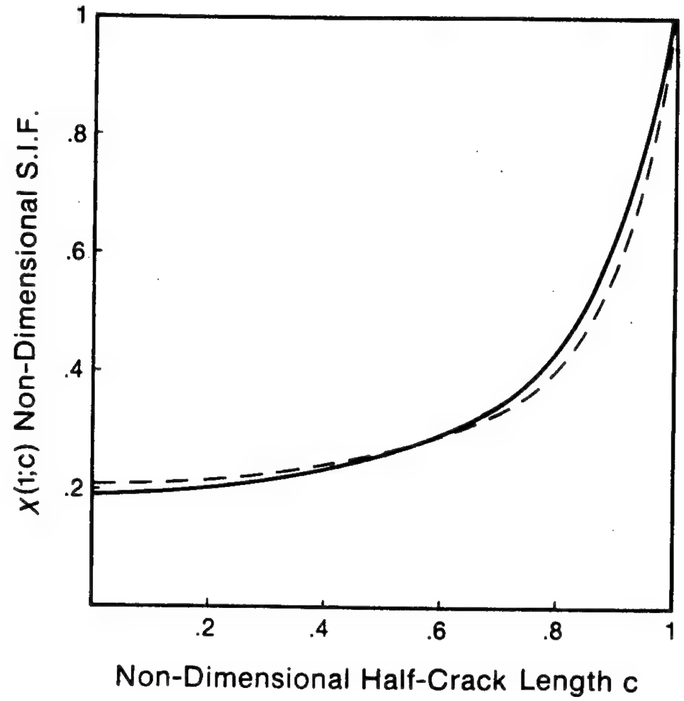


Figure 7. Non-dimensional stress intensity factors X(1) vs. c. Non-linear results (solid lines) and linear values (dashed lines). c is the ratio of the length of the central crack to the length of the craze.

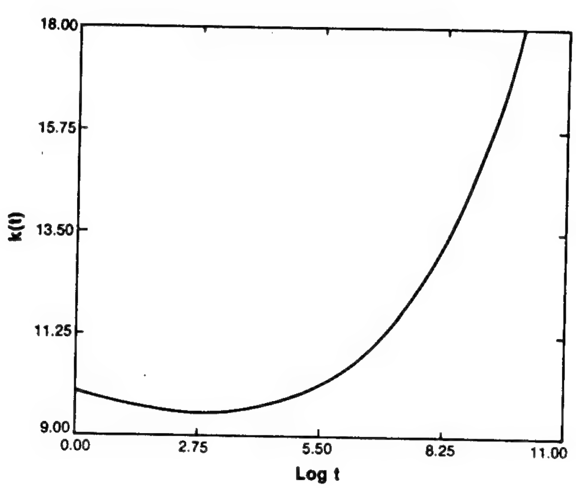


Figure 8. Variations of k(t) vs. log-time in the second case. k(t) is the time-dependent modulus of the craze.

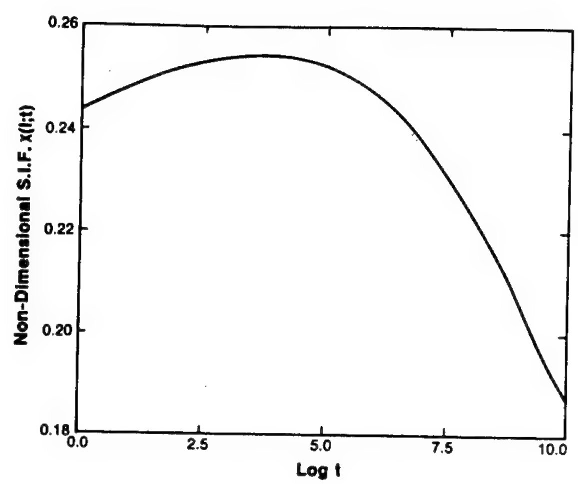


Figure 9. Variation of the stress intensity factor X(1;t) vs. log time when the craze modulus tightens with time and the bulk modulus relaxes with time.

Table 1. Summary of Hygrothermal Conditions and Data Obtained by Harper [9] and in Present Work (Jackson).

T (F)	RH (%)	Conditioning History	Harper	Jackson
77	13	Saturating from dry	■	□*
77	75	Saturating from dry	■	□*
77	95	Saturating from dry	■	□*
130	75 & 0	Saturated at 75% RH	■	
		Dried at 0% RH		□*
130	95 & 0	Saturated at 95% RH	■	
		Dried at 0% RH		□*
130	95 & 0	9 day cycling between 95% & 0% RH [1]	■	□
130	95 & 0	16 day cycling between 95% & 0% RH [1]	■	□
150	95 & 0	9 day cycling between 95% & 0 RH [2]		□

* Continued by Jackson, but conditioning still ongoing.

[1] Starting from initially saturated condition.
[2] Starting from initially dry condition.

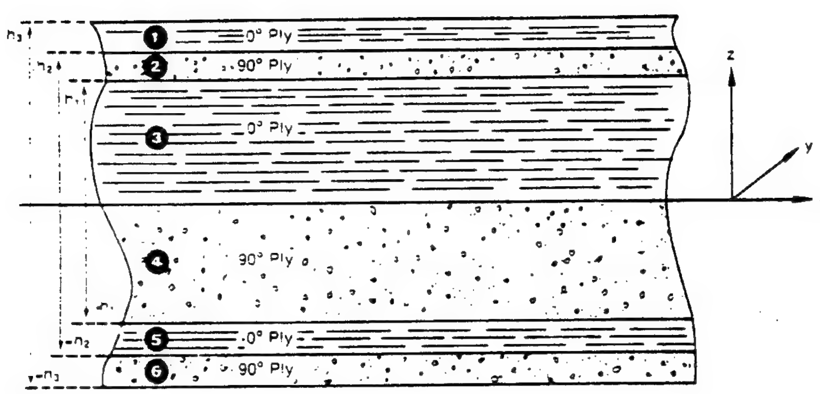


Figure 10. Geometry of anti-symmetric cross-ply laminates.

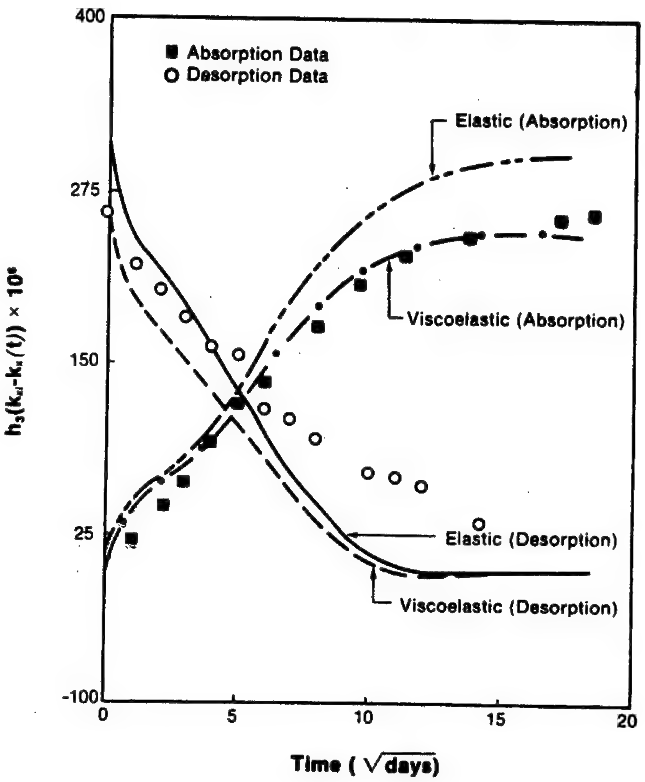
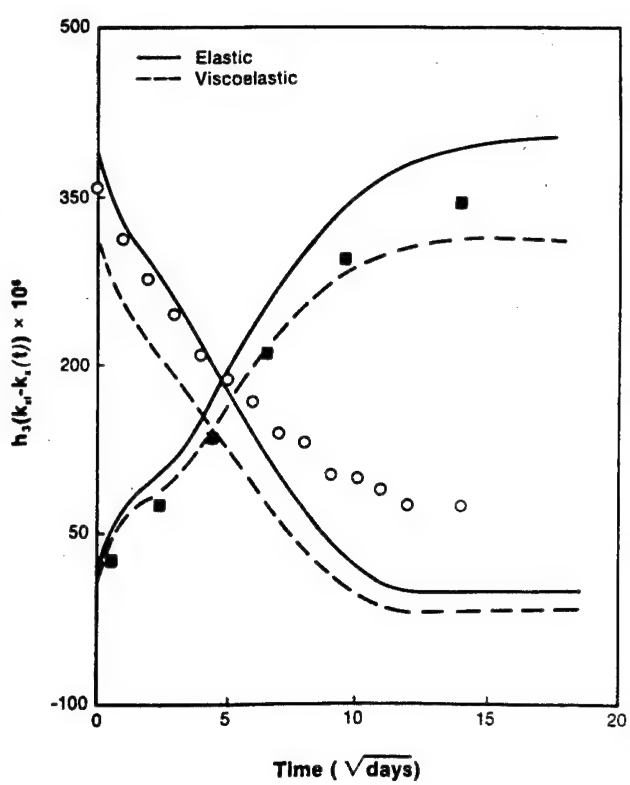


Figure 11. Time-dependent curvature change of $(0/90/0_4/90_4/0/90)_T$ AS4/3502 graphite/epoxy laminates at 130°F. Absorption data was obtained at 75% relative humidity.

Figure 12. Time dependent curvature change of $(0/90/0_4/90_4/0/90)_T$ AS4/3502 graphite/epoxy laminates at 130°F. Absorption data was obtained at 95% relative humidity.



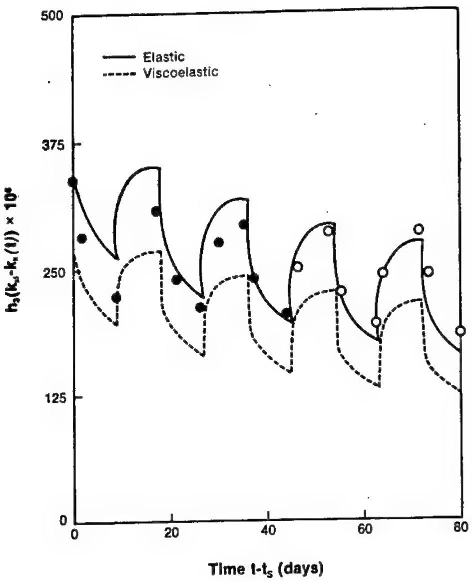


Figure 13. Time-dependent curvature change of $(0/90/0_4/90_4/0/90)_T$ AS4/3502 graphite/epoxy laminates during cyclic exposure to 0% and 95% R.H. at 130°F, with cycle interval of 9 days.

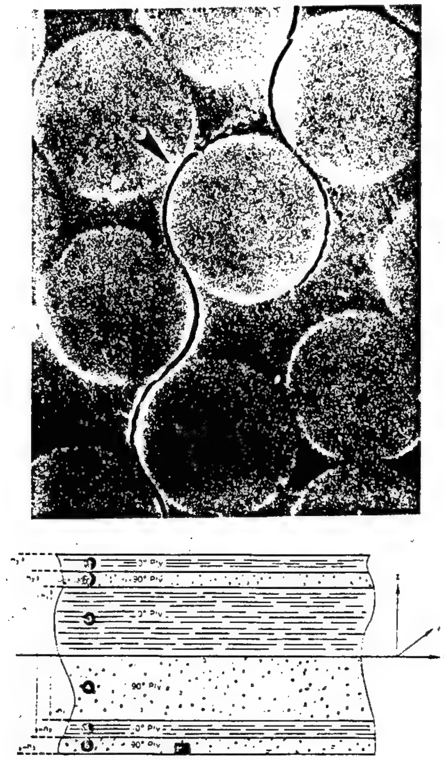


Figure 14. Vertical microcrack starting just underneath the surface of a laminate that was initially saturated, (95% R.H.), and then cycled at 9 day intervals between 95% and 0% R.H. at 130°F, 4500X.

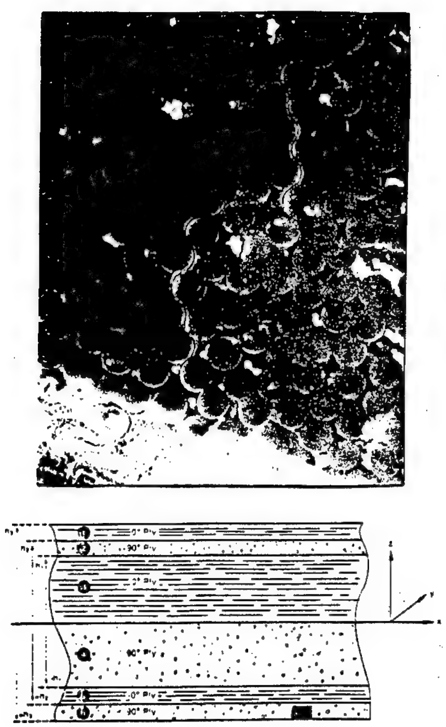


Figure 15. Vertical microcrack originating at the surface of a laminate that was initially saturated (95% R.H. at 130°F), then cycled at 9 day intervals between 0% and 95% R.H. at 130°F, 1000X.

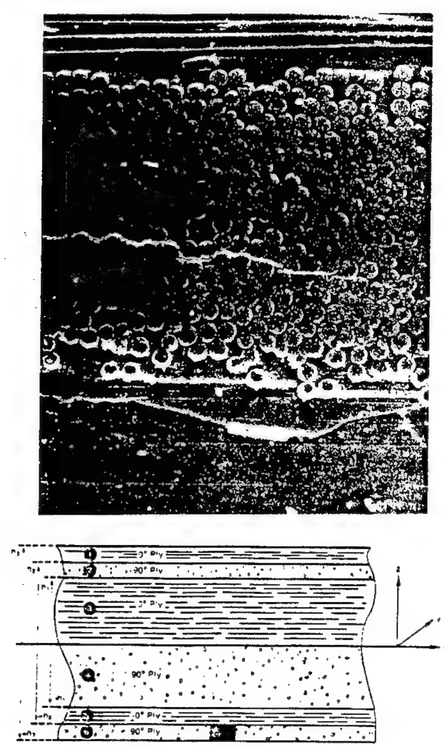


Figure 16. A horizontal microcrack that runs parallel to the surface of an initially dry laminate that was cycled at 9 day intervals between 95% and 0% R.H. at 150°F, 450X.

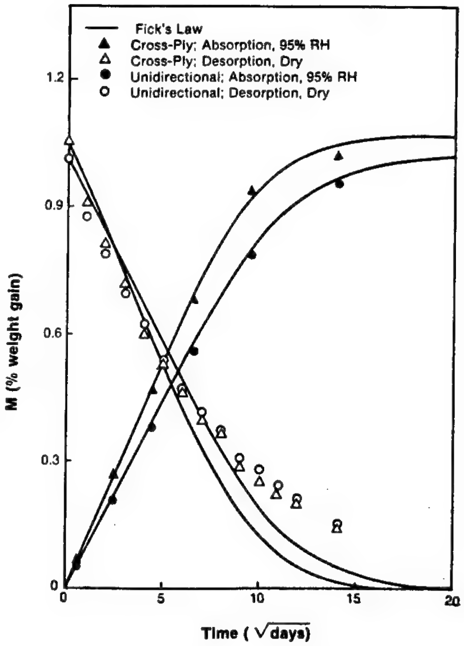


Figure 17. Absorption and desorption for unidirectional and cross-ply specimens at 130°F, 450X.

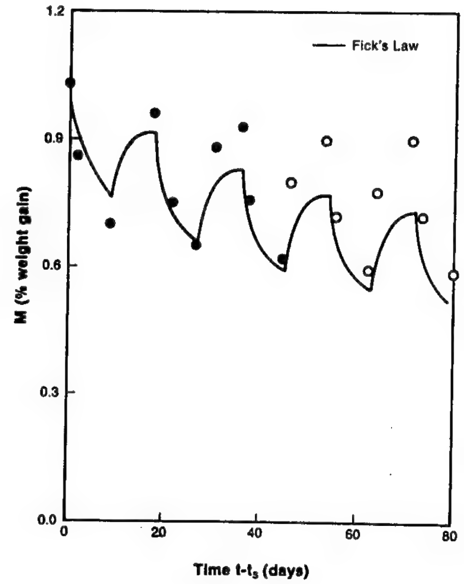


Figure 18. Moisture content (in % weight gain) during cyclic exposure to 0% and 95% relative humid-itits at 130°F, with cycle interval of 9 days.

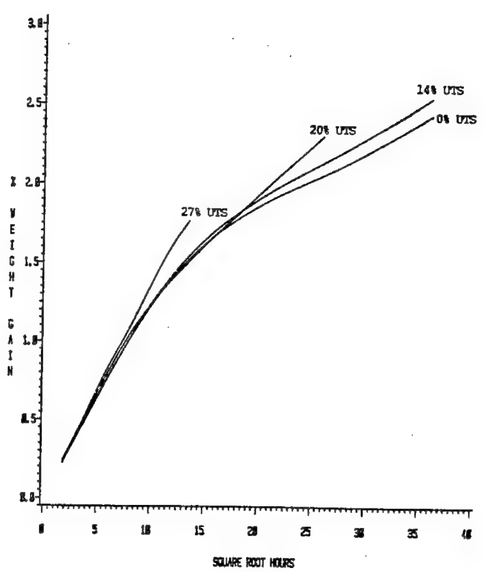


Figure 19. Percent weight gain in tensile coupons of F155 composite material subjected to different stress levels during exposure to 41°C and 93% R.H. Coupons loaded transversely. UTS = Ultimate Tensile Strength (in transverse direction).

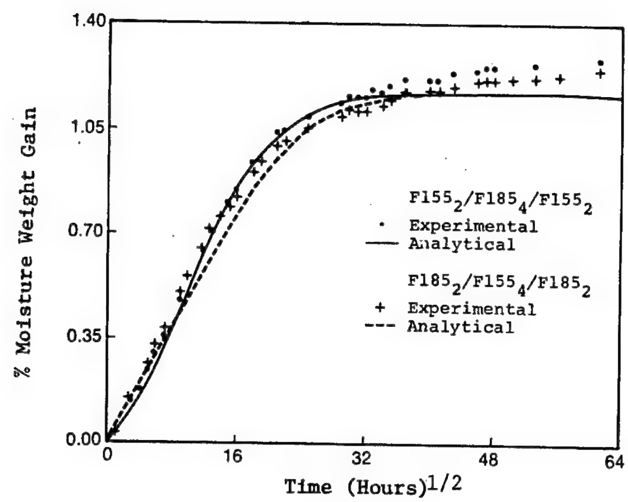


Figure 20. Moisture weight gain in hybrid composite specimens: 49°C, 95% R.H.

COMPUTER MODELING OF THE COMPOSITE THERMAL CURE CYCLE: AN OVERVIEW

RONALD A. SERVAIS

UNIVERSITY OF DAYTON
DAYTON, OHIO 45469

ABSTRACT

Composite materials, consisting of resin and fiber constituents, are used because selected properties such as low density, high strength, ability to withstand high temperatures, and resistance to environmental effects compare favorably to alternative materials. A large variety of composites have been developed, each with special characteristics. The typical fabrication procedure consists of weaving a cloth from the fibers, impregnating the cloth with resin, stacking the impregnated cloth to some desired thickness, and then applying both heat and pressure according to a predetermined plan (called a cure cycle) to form the composite product.

The properties of the constituents and the cure cycle establish the characteristics and the quality of the product. Historically, the cure cycle was determined from a combination of experience and trial and error; this approach was usually adequate because of the large processing window available. Now, however, the product quality requirements have increased as well as an evolutionary improvement in composite constituents; this has led to a reduction in the processing window, hence less room for error. The ability to predict the behavior of the composite material during the curing cycle, based on constituent properties, would reduce fabrication costs and insure product quality. In addition, a process curing model would clearly be a valuable tool for the materials development engineer.

In general, the process curing models consist of several components; these include the constraints or restrictions (the assumptions), the model itself (that is, the mathematical descriptions of the critical phenomena, sometimes called sub-models, and resulting in a system of equations), the numerical technique utilized to solve the system of equations, and the computer program or code which represents the application of the previous components. In addition, material properties, such as density and thermal conductivity, along with boundary conditions, including initial temperatures and water vapor along with the processing temperature and pressure histories, feed into the computer program. At this point, sufficient information and descriptions of the process are available in order to predict a solution which would typically include temperature-time distributions, degree of cure, degree of compaction, and related results. The model can now be used to easily and cost-effectively evaluate various curing cycles in order to determine the impact on the quality of the composite. Presumably, the results can be verified by comparing with actual curing data such as internal temperature distribution during the curing process as well as the degree of cure.

Two computer programs are being used to model the composite curing process. These are the code developed by George S. Springer (1) and a proprietary code which was developed by Lockheed-Georgia (2). The Springer code is called "CURE" and the Lockheed-Georgia code is called "ROAST".

The use of both computer programs is similar. In each case, material properties (such as density, thermal conductivity, the heat release due to the reaction, and related properties), geometric data (laminate dimensions), as well as the boundary conditions (autoclave temperature history, for example) are specified. The computer program is then used to solve the energy equation in order to predict the internal temperature distribution as a function of time within the curing composite. In turn, the degree of cure, viscosity, compaction, and related parameters can be predicted and then used to evaluate the cure cycle (boundary conditions).

The degree of sophistication is quite different for the two programs. CURE is limited to a one-dimensional heat transfer analysis but incorporates fiber and

resin (constituent) properties. ROAST can accommodate multi-dimensional heat transfer at the expense of constituent property data; it presumes that the composite consists of homogeneous layers of constant property materials. Neither approach represents an appropriate starting point for a (multi-dimensional) mechanics analysis for determining internal stresses and shrinkage associated with the cure and cool-down cycle. Both programs require the appropriate material properties in order to actually investigate a cure cycle.

Several versions of CURE are available including AMDAHL, PRIME 850, and IBM PC versions; a separate HP Plotter graphics routine is also available for the IBM PC version. At this time, ROAST is considered to be a proprietary computer program by Lockheed-Georgia and the source code is not available.

The two available computer codes can be considered to include several sub-models such as the kinetics and the viscosity sub-models. It should be noted that the mathematical equations which represent the various sub-model phenomenological behavior can readily be altered in the computer codes. The viscosity relations which are used in the two codes are different but appear to adequately represent the experimental data. The kinetics expressions also appear to be adequate.

The flow and the void modeling components are clearly related. The existing models do not adequately predict the actual void growth or behavior. Internal pressure distributions are required to verify the flow/void sub-models; the pressure data is not available either. Before cure cycle models can be used to predict void formation/movement and resin flow, appropriate models must be developed and then verified using internal pressure data which must be experimentally obtained.

A case consisting of 64 plies of Hercules AS/3501-6 has been run on the Springer code to illustrate typical output. The input temperature and pressure cure histories are shown along with the predicted viscosity and degree of cure in the middle layer of the composite as a function of time as well as the overall predicted compaction of the composite. This particular case required less than ten minutes running time on an IBM PC. It should be noted that only AS/3501-6 properties are available for the Springer code at this time; an extensive effort will be required to extend this property data base.

Some of the special problems associated with process cure cycle models can be identified. Codes must be manageably small in order to be compatible with microcomputers, accessible to engineers, and run cases in real time for process control. The data base must be extended to include a range of materials of interest. The heat transfer analysis must accommodate complex geometric shapes, including compatibility with mechanics analyses. The flow/void descriptions must be extended in order to be able to investigate their effects on product quality. The submodels, the extended data base, and the final code predictions must be verified in order to develop confidence in the modeling approach. For the models to be useful in a production environment, appropriate sensors for monitoring internal in-situ composite behavior (probably including pressure, compaction, and void measurements) must be developed.

REFERENCES

1. Springer, G. A. and Loos, A. C., "Curing of Graphite/Epoxy Composites", AFWAL-TR-4040, March, 1983.

2. Private communication with Ancil Kays, Lockheed-Georgia, Marietta, Georgia 30063.

COMPUTER MODELING

OF THE COMPOSITE THERMAL CURE CYCLE:

AN OVERVIEW

RONALD A. SERVAIS

UNIVERSITY OF DAYTON

<u>COMPOSITE MATERIALS</u>

MATRIX/FIBER

HEAT/PRESSURE CURE

AUTOCLAVE/PRESS

HEAT RELEASE

VOLATILES

<u>BACKGROUND</u>

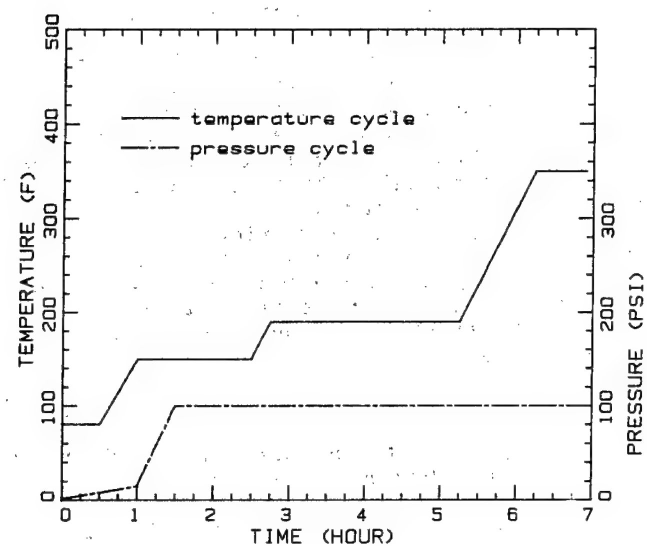
COMPOSITE MATERIALS

THERMAL/FORCE CURE CYCLES

CYCLE DETERMINATION

HEAT BALANCE

PRODUCT QUALITY FACTORS



TEMPERATURE AND PRESSURE CURE CYCLES

CYCLE DETERMINATION

EXPERIENCE

TRIAL AND ERROR

PRODUCT QUALITY FACTORS

DEGREE OF CURE

PERCENT COMPACTION

VOID CONTENT/DISTRIBUTION

FIBER/RESIN ADHESION

SHRINKAGE

INTERNAL STRESSES

ENERGY BALANCE

$$\rho C \frac{\partial T}{\partial t} = k \frac{\partial^2 T}{\partial x^2} + S$$

ρ = density (gm/cm^3)

C = specific heat ($cal/gm\text{-}^{o}K$)

T = temperature (^{o}K)

t = time (sec)

k = thermal conductivity ($cal/cm\text{-}sec\text{-}^{o}K$)

x = vertical position (cm)

S = heat source ($cal/cm^3\text{-}sec$)

COMPUTER MODEL COMPONENTS

CONSTRAINTS (Assumptions and Restictions)

PROPERTIES (Thermodynamic, Transport, Mechanical Data)

MODEL (Phenomenological Description)

NUMERICAL METHOD (Solution Technique)

COMPUTER PROGRAM (Code)

BOUNDARY CONDITIONS (Initial and Historical Environment)

VERIFICATION (Experimental and Mathematical Confirmation)

CONSTRAINTS

ONE-/MULTI-DIMENSIONAL HEAT TRANSFER

HOMOGENEOUS/CONSTITUENT PROPERTIES

TYPES OF BOUNDARY CONDITIONS

CLASSES OF MATERIALS

EASE OF USE/UNDERSTANDING

COMPUTER HARDWARE/TIME REQUIREMENTS

GRAPHICS OUTPUT

MODELS

KINETICS

VISCOSITY

FLOW

VOID FORMATION/MOVEMENT

INTERNAL STRESSES

THERMAL EXPANSION

GEL POINT

PROPERTIES

KINETIC (reaction parameters)

THERMODYNAMIC (density, specific heat)

TRANSPORT (thermal conductivity, viscosity)

CHEMICAL (composition, water vapor, volatiles)

PHYSICAL (ply thickness and orientation, dimensions, permeability, compaction)

NUMERICAL METHODS

FINITE DIFFERENCE METHOD

FINITE ELEMENT METHOD

ELECTRICAL ANALOGY METHOD

COMPUTER PROGRAMS

CURE (SPRINGER)

ROAST (LOCKHEED)

VERIFICATION

COMPOSITION

UNIFORMITY

TEMPERATURE DISTRIBUTION DURING CURE

PRESSURE DISTRIBUTION DURING CURE

VOID SIZE AND DISTRIBUTION

COMPACTION

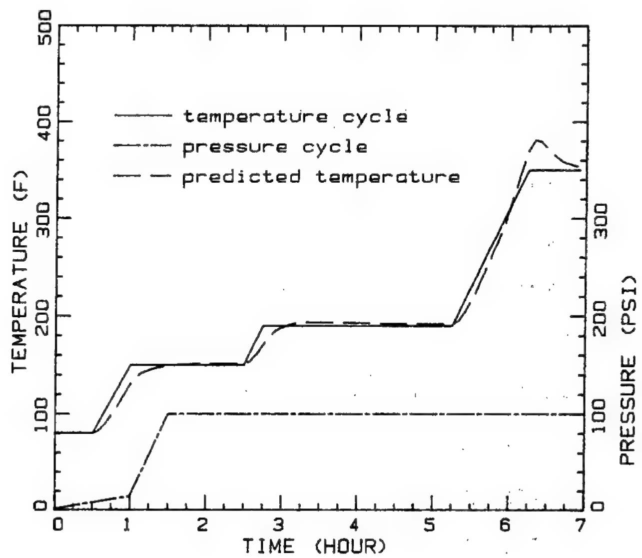
INTERNAL CRACKS

MATHEMATICAL CONVERGENCE

BOUNDARY CONDITIONS

INITIAL CONDITIONS (Temperature,
Water Vapor,
Void Distribution)

BOUNDARY CONDITIONS (Temperature History,
Pressure History)



PREDICTED INTERNAL TEMPERATURE FOR 64-PLY. AS/3501-6

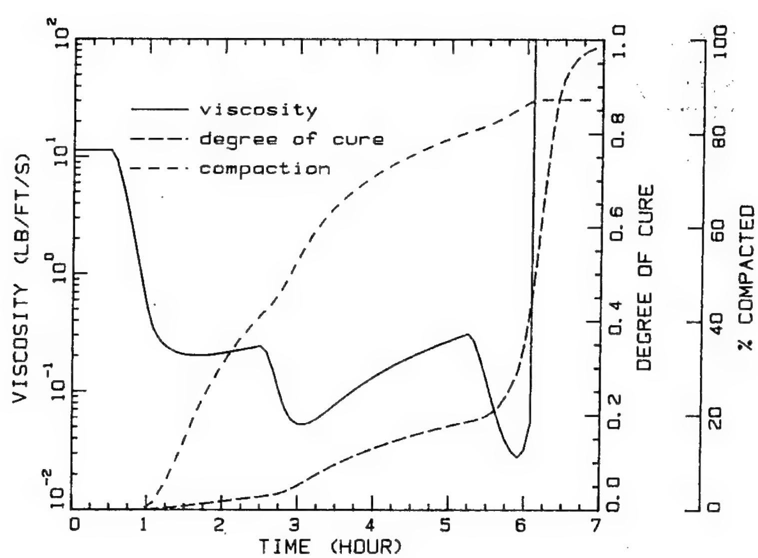
SPECIAL PROBLEMS

FLEXIBILITY

PROPERTY DETERMINATION

EASE OF USE

COMPUTER REQUIREMENTS



PREDICTED VISCOSITY, DEGREE OF CURE, AND TOTAL COMPACTION

CURRENT STATUS - CURE PROCESS CODE COMPARISON

CAPABILITY	CURE	ROAST
ONE-DIMENSIONAL HEAT TRANSFER	YES	YES
TWO-DIMENSIONAL HEAT TRANSFER	NO	YES
LANGUAGE	FORTRAN	FORTRAN
CODE LENGTH	LARGE	SMALL
COMPUTER RUN TIME	SMALL	LARGE
NUMERICAL TECHNIQUE	FINITE DIFFERENCES	ELECTRICAL ANALOGY
PROPERTIES	CONSTITUENT	COMPOSITE
KINETIC MODEL	ARBITRARY	ARBITRARY
VISCOSITY MODEL	ARBITRARY	ARBITRARY
FLOW MODEL	CRUDE	IGNORED
VOID MODEL	CRUDE	IGNORED
INPUT VARIABLES	MANY	FEW
MECHANICS EASILY ADDED	NO	NO

FUTURE

OPTIMIZATION

CURE AND POST-CURE STRESSES

INSTRUMENTATION

PROCESS CONTROL

DAMAGE ACCUMULATION IN PLY-TERMINATION SPECIMENS

D. A. Ulman
H. R. Miller

General Dynamics Fort Worth Division

ABSTRACT

The increased usage of advanced composite materials on aircraft structures necessitates the creation of effective methodologies for the prediction of life in these materials. The establishment of two such life prediction procedures is the objective of this AFWAL/FDL-sponsored program. The first of these life prediction methodologies is founded upon the experimentally documented damage accumulation process. The other procedure for life prediction is based upon the reduction in material stiffness that is associated with the development of damage in composite laminates. The required information is being generated through an extensive experimental program. The damage accumulation process and subsequent stiffness change are being documented under a wide variety of load conditions. The chronology and correlations between the observed damage states and laminate stiffness provides a foundation for the understanding of the basic mechanisms that control the strength and life of composite aircraft components.

The damage accumulation process in thick graphite/epoxy ply-termination coupons has been extensively documented. Stiffness measurements and various non-destructive evaluation techniques were used to study damage growth and laminate property degradation under a wide variety of test conditions. The effects of stress-level, stress-range, and loading mode on the damage accumulation and stiffness degradation processes were investigated through constant amplitude fatigue tests. The resulting stiffness change and damage development data were used in the development of life-prediction procedures.

The damage accumulation observed in ply-termination specimens consisted of transverse - matrix cracking and delaminations. Transverse matrix cracks were observed in all test conditions with tensile load excursions. The larger the tensile stress the sooner the development of the transverse crack saturation spacing. Delamination growth dominated the later stages of fatigue life. Stress-ratio, stress-level, and stress-history were also found to make significant contributions to damage development.

Stiffness change, like damage accumulation, was sensitive to the fatigue loading condition. Correlations between laminate damage state and the laminate stiffness loss were observed. Thus, both the damage state and stiffness change can be used as valid metrics for life-prediction models.

The two life-prediction procedures are based upon the assumption that equivalent damage-metrics produce the same remaining life regardless of the preceding load history. The damage development and stiffness degradation data from constant amplitude fatigue test were used to develop the life-prediction procedures. Two stage spectrum fatigue test were performed and the experimental results compared with the predicted specimen lives.

This research was performed under the Air Force sponsored program Damage Accumulation in Composites, F33615-81-C-3226. George Sendeckyj of AFWAL/FIBEC is the Air Force Project Engineer.

DAMAGE ACCUMULATION IN PLY-TERMINATION SPECIMENS

D. A. ULMAN
H. R. MILLER

GENERAL DYNAMICS
FORT WORTH DIVISION

AGENDA

- INTRODUCTION
- LIFE-PREDICTION PROCEDURE DEVELOPMENT
  - STIFFNESS CHANGE BASED
  - DAMAGE STATE BASED
- SPECTRUM LIFE-PREDICTION
- SUMMARY

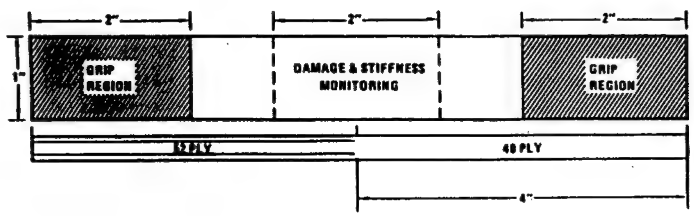
OBJECTIVE

- TO DEVELOP LIFE-PREDICTION PROCEDURE FOR COMPOSITE LAMINATES BASED UPON EXPERIMENTALLY OBSERVED DAMAGE METRICS

TECHNICAL APPROACH

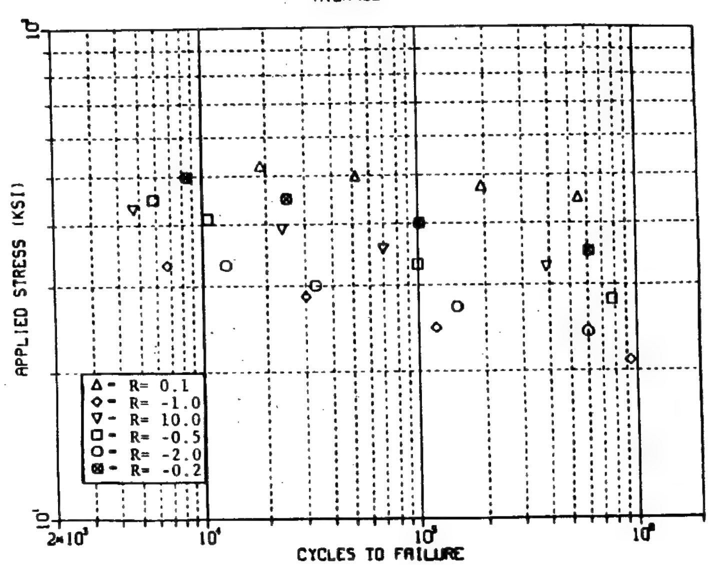
- EXTENSIVELY TEST A SINGLE LAMINATE CONFIGURATION
- INCREMENTALLY MONITOR THE LAMINATE STIFFNESS AND DAMAGE STATE
- DEVELOP LIFE-PREDICTION PROCEDURES
- PREDICT THE LIFE OF THE LAMINATE UNDER SPECTRUM FATIGUE LOADING

PLY-TERMINATION SPECIMEN



Section	Family	Stacking Sequence *
Thin	(25/50/25)	$[(0/+/90/-)_s\ (0/-/90/+)_s\ (0/+/90/-)_s]_s$
Thick	(25/50/25)	$[(0/+/90/-)_s\ (0/-/90/+/0)_s\ (0/+/90/-)_s]_s$ └ TERMINATED PLIES

AVERAGE S-N



TEST MATRIX

TEST TYPE	LOADING	STRESS RATIO	TEST CONDITIONS	NO. REPLICATES	NO. SPEC	TOTAL
Static	Tension	–	Rapid Ramp to Failure	5	5	20
			Same With Damage Monitoring	5	5	
	Compression	–	Rapid Ramp to Failure	5	5	
			Same With Damage Monitoring	5	5	
Fatigue	Constant Amplitude	0.1	Four Different Stress Levels	5	20	120
		–1	Four Different Stress Levels	5	20	
		10	Four Different Stress Levels	5	20	
		–0.5	Four Different Stress Levels	5	20	
		–2.0	Four Different Stress Levels	5	20	
		–0.2	Four Different Stress Levels	5	20	
Spectrum	Two-Stage and Flight-by-Flight Spectrums	–	At Least 10 Flight-by-Flight Spectrums Out of 20 Total	5	50	100
			Balance to Be Simple Compression-dominated Two-Stage Spectrums	5	50	
						240

LIFE PREDICTION METHODOLOGIES

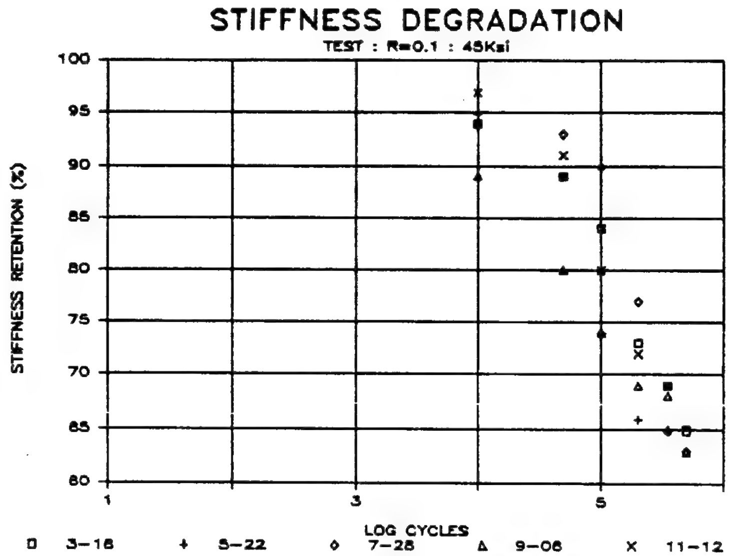
- CONCEPTS/ASSUMPTIONS -

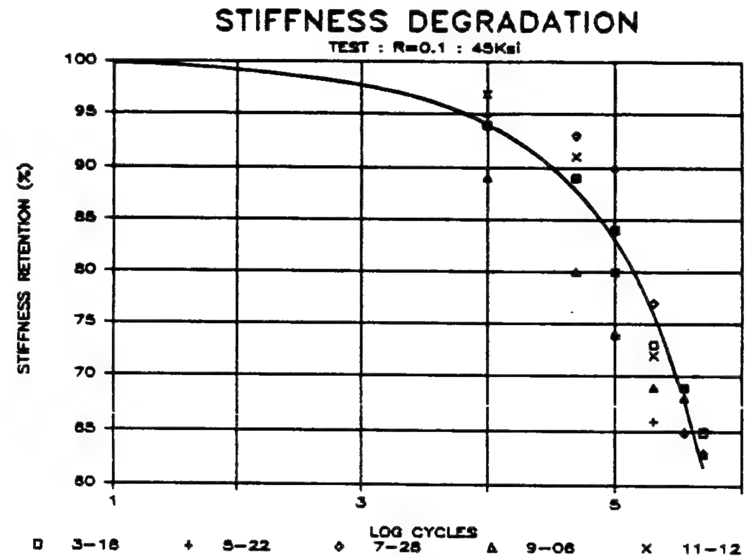
o Equivalent Damage States and Stiffness Retentions Can Be Related to Used Remaining Life

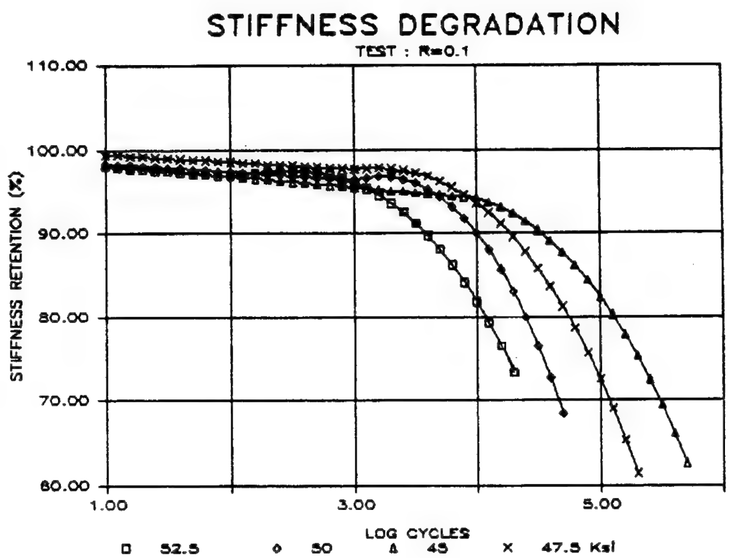
o Stiffness Change is an Effective Measure of Mechanical Property Degradation

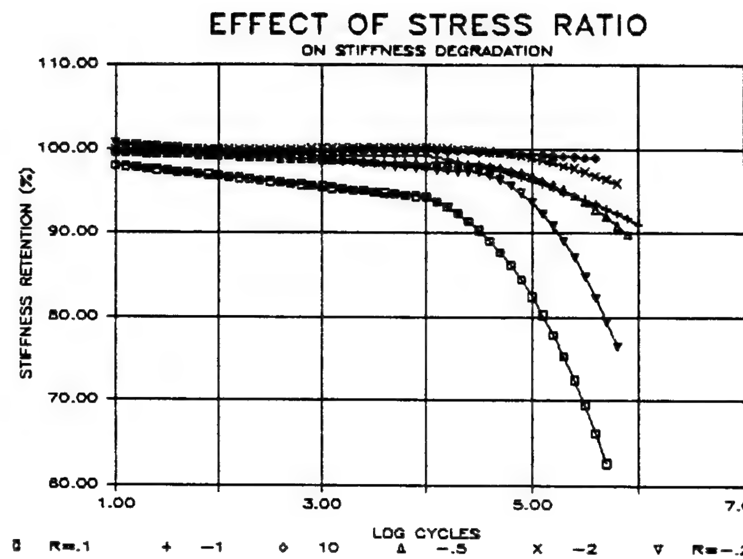
o Damage Accumulation is Systematic and Progressive

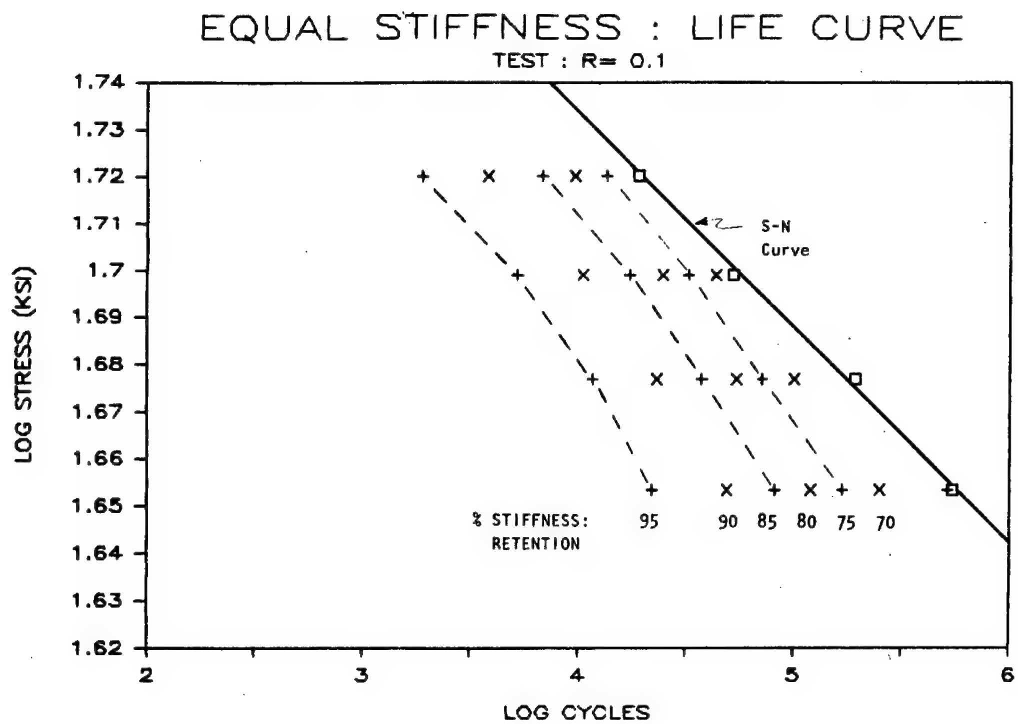
o Stiffness Degradation Relates to Damage Development

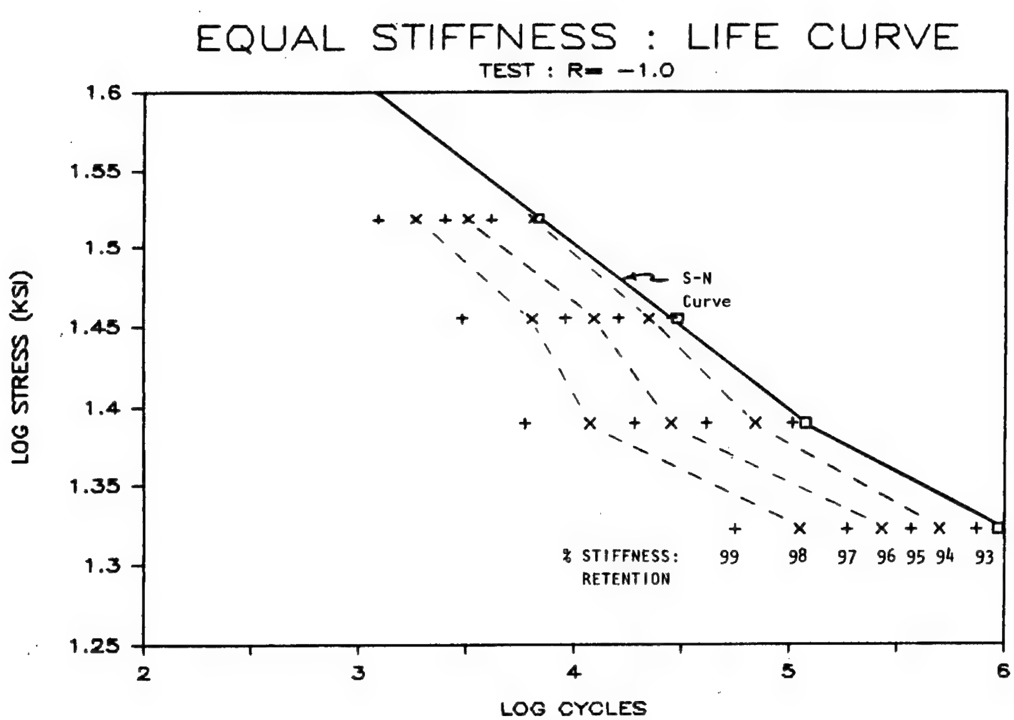

STIFFNESS DEGRADATION
TEST : R=0.1 : 45Ksi
STIFFNESS RETENTION (%)
LOG CYCLES
□ 3-16 + 5-22 ◇ 7-28 △ 9-06 × 11-12


STIFFNESS DEGRADATION
TEST : R=0.1 : 45Ksi
STIFFNESS RETENTION (%)
LOG CYCLES
□ 3-16 + 5-22 ◇ 7-28 △ 9-06 × 11-12

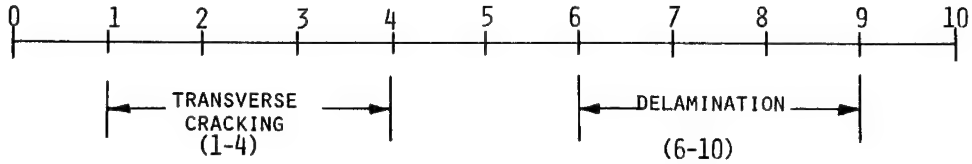

STIFFNESS DEGRADATION
TEST : R=0.1
STIFFNESS RETENTION (%)
LOG CYCLES
□ 52.5 ◇ 50 △ 45 × 47.5 Ksi

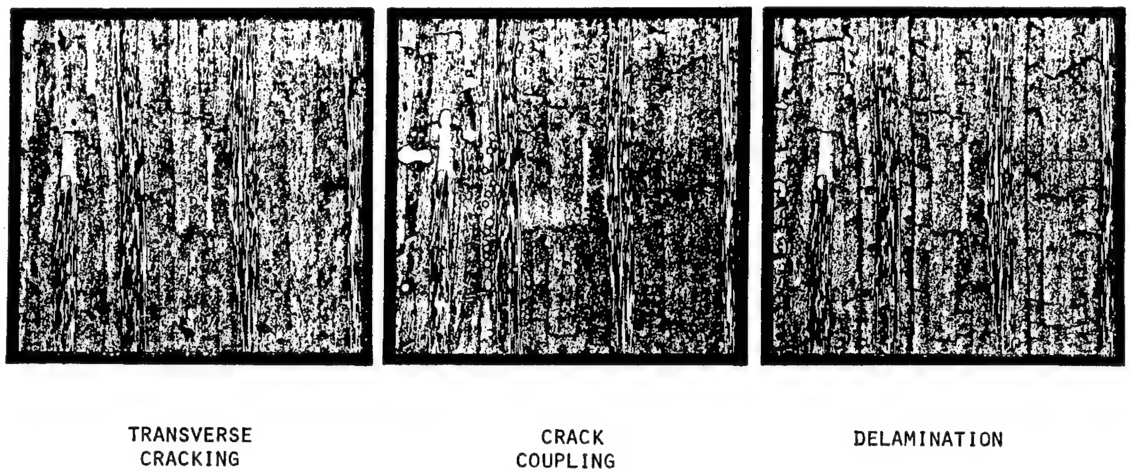

EFFECT OF STRESS RATIO
ON STIFFNESS DEGRADATION
STIFFNESS RETENTION (%)
LOG CYCLES
□ R=.1 + -1 ◇ 10 △ -.5 × -2 ▽ R=-.2

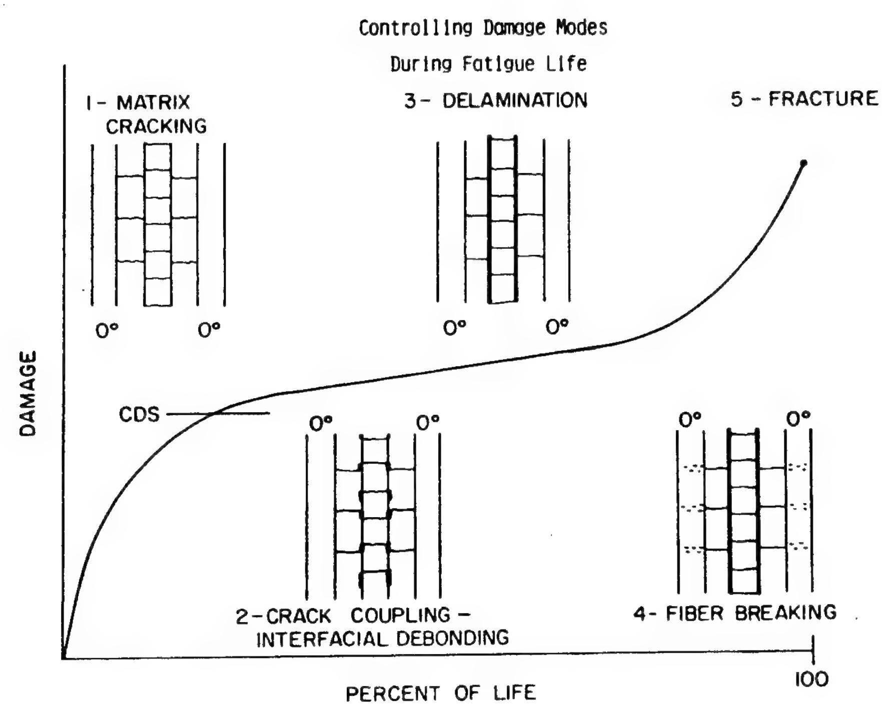

EQUAL STIFFNESS : LIFE CURVE
TEST : R= 0.1
LOG STRESS (KSI)
LOG CYCLES
S-N Curve
% STIFFNESS: RETENTION
95
90
85
80
75
70
1.74
1.73
1.72
1.71
1.7
1.69
1.68
1.67
1.66
1.65
1.64
1.63
1.62
2
3
4
5
6

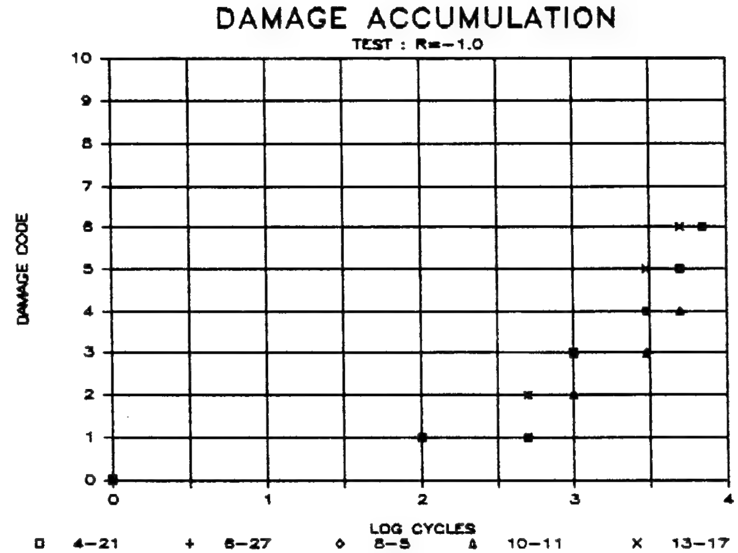

EQUAL STIFFNESS : LIFE CURVE
TEST : R= -1.0
LOG STRESS (KSI)
LOG CYCLES
S-N Curve
% STIFFNESS: RETENTION
99
98
97
96
95
94
93
1.6
1.55
1.5
1.45
1.4
1.35
1.3
1.25
2
3
4
5
6

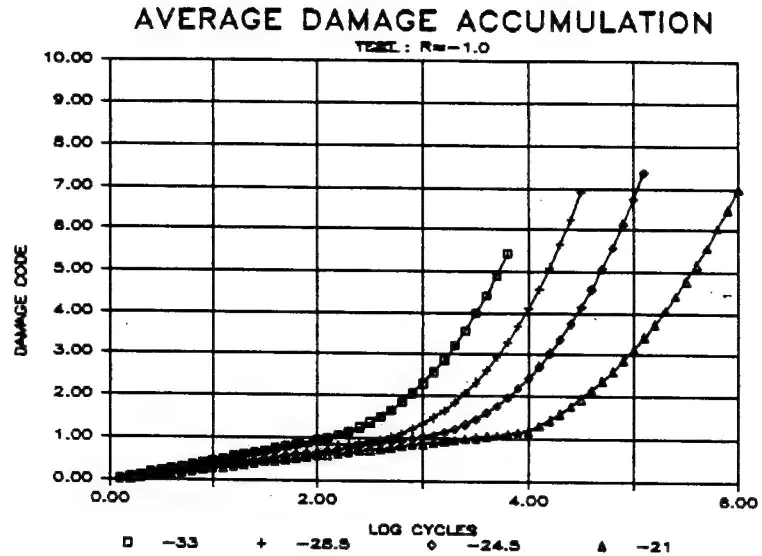
DAMAGE CODE

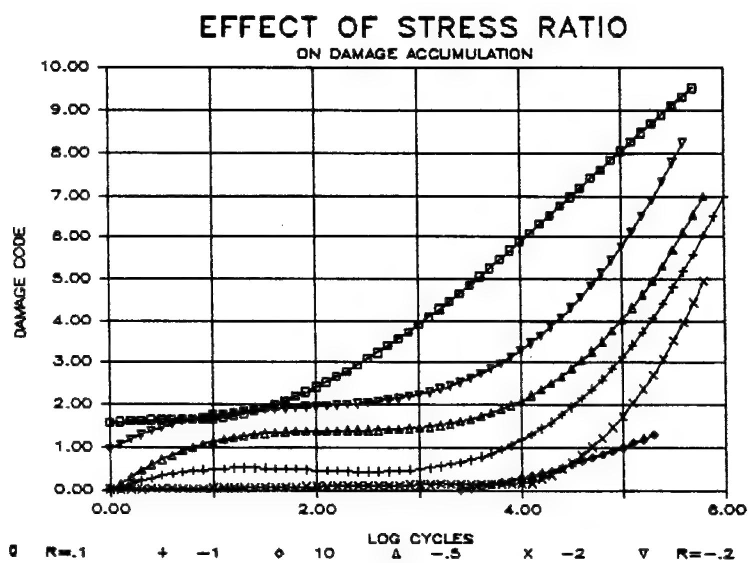

0 1 2 3 4 5 6 7 8 9 10
TRANSVERSE CRACKING (1-4)
DELAMINATION (6-10)

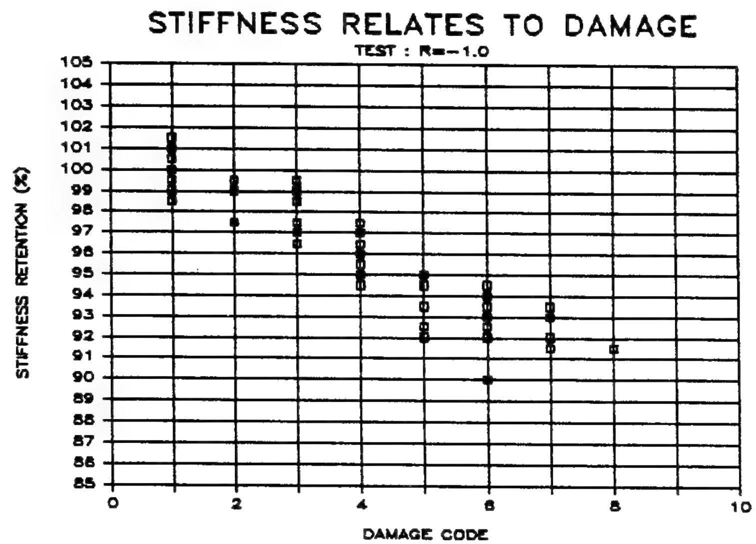

TRANSVERSE CRACKING
CRACK COUPLING
DELAMINATION

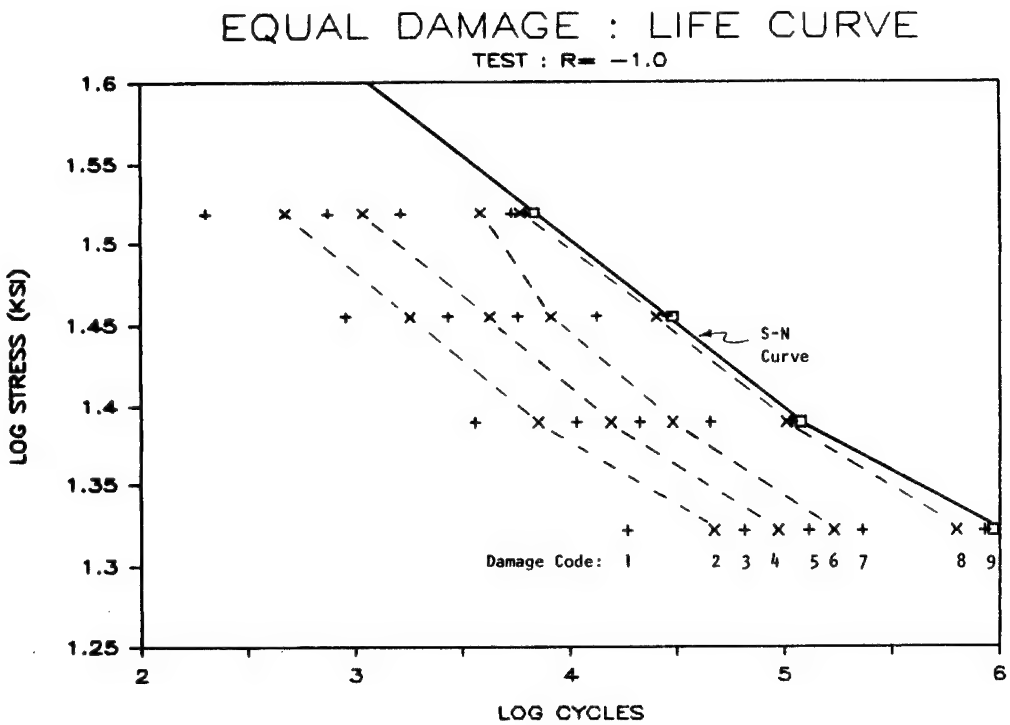

Controlling Damage Modes
During Fatigue Life
1- MATRIX CRACKING
3- DELAMINATION
5 - FRACTURE
0°
DAMAGE
CDS
2-CRACK COUPLING - INTERFACIAL DEBONDING
4- FIBER BREAKING
PERCENT OF LIFE
100

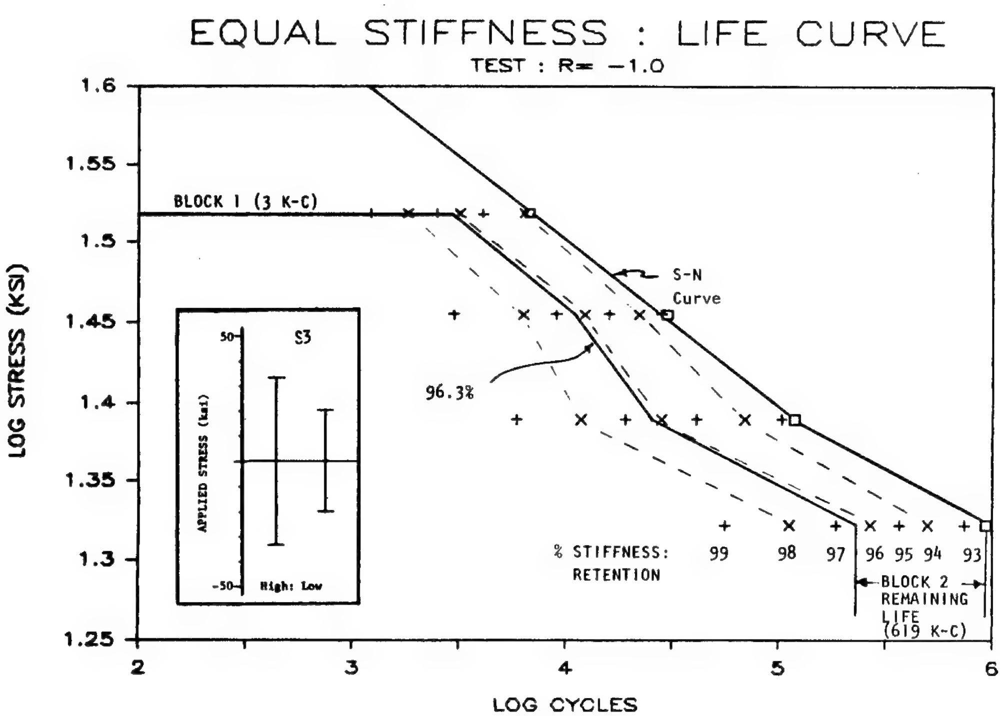

DAMAGE ACCUMULATION
TEST : R=-1.0
DAMAGE CODE
LOG CYCLES
4-21
6-27
5-5
10-11
13-17


AVERAGE DAMAGE ACCUMULATION
R=-1.0
DAMAGE CODE
LOG CYCLES
-33
-28.5
-24.5
-21


EFFECT OF STRESS RATIO
ON DAMAGE ACCUMULATION
DAMAGE CODE
LOG CYCLES
R=.1
-1
10
-.5
-2
R=-.2

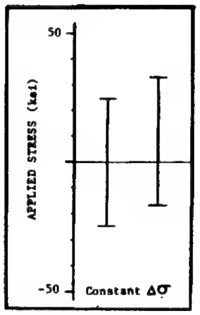

STIFFNESS RELATES TO DAMAGE
TEST : R=-1.0
STIFFNESS RETENTION (%)
DAMAGE CODE


EQUAL DAMAGE : LIFE CURVE
TEST : R= -1.0
LOG STRESS (KSI)
LOG CYCLES
S-N
Curve
Damage Code: 1 2 3 4 5 6 7 8 9
1.6
1.55
1.5
1.45
1.4
1.35
1.3
1.25
2
3
4
5
6


EQUAL STIFFNESS : LIFE CURVE
TEST : R= -1.0
LOG STRESS (KSI)
LOG CYCLES
BLOCK 1 (3 K-C)
S-N
Curve
96.3%
S3
APPLIED STRESS (ksi)
50
-50
High: Low
% STIFFNESS:
RETENTION
99 98 97 96 95 94 93
BLOCK 2
REMAINING
LIFE
(619 K-C)
1.6
1.55
1.5
1.45
1.4
1.35
1.3
1.25
2
3
4
5
6

TWO-STAGE SPECTRA LIFE PREDICTIONS

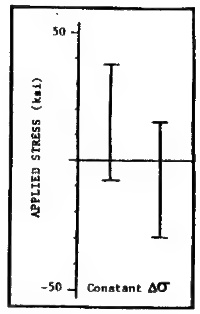
TEST S5



C.A. LIFE BLOCK 1	108	K-C
C.A. LIFE BLOCK 2	99.5	K-C
BLOCK 1 CYCLES	50	K-C

	PREDICTION	OBSERVED
DAMAGE STATE (END OF BLOCK 1)	7.1	6.0
STIFFNESS (%) (END OF BLOCK 1)	94.7	95.0
REMAINING LIFE		
DAMAGE (K-C)	71.5	149
STIFFNESS (K-C)	75.0	149

TEST S6

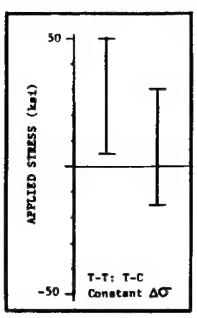


C.A. LIFE BLOCK 1	241	K-C
C.A. LIFE BLOCK 2	33.2	K-C
BLOCK 1 CYCLES	100	K-C

	PREDICTION	OBSERVED
DAMAGE STATE (END OF BLOCK 1)	8.5	8.7
STIFFNESS (%) (END OF BLOCK 1)	89.2	90.6
REMAINING LIFE		
DAMAGE (K-C)	0	0.01
STIFFNESS (K-C)	0	0.01

TWO-STAGE SPECTRA LIFE PREDICTIONS

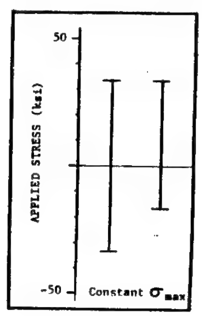
TEST S9



C.A. LIFE BLOCK 1	52.0	K-C
C.A. LIFE BLOCK 2	327	K-C
BLOCK 1 CYCLES	10	K-C

	PREDICTION	OBSERVED
DAMAGE STATE (END OF BLOCK 1)	8.4	8.8
STIFFNESS (%) (END OF BLOCK 1)	92.0	93.6
REMAINING LIFE		
DAMAGE (K-C)	104.4	36.5
STIFFNESS (K-C)	34.0	36.5

TEST S10

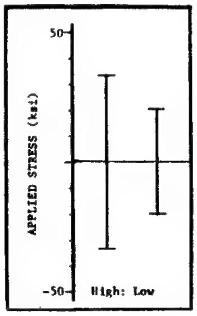


C.A. LIFE BLOCK 1	6.75	K-C
C.A. LIFE BLOCK 2	99.5	K-C
BLOCK 1 CYCLES	4.5	K-C

	PREDICTION	OBSERVED
DAMAGE STATE (END OF BLOCK 1)	6.5	6.2
STIFFNESS (%) (END OF BLOCK 1)	94.8	95.0
REMAINING LIFE DAMAGE (K-C) STIFFNESS (K-C)	79.5	93.7

TWO-STAGE SPECTRA LIFE PREDICTIONS

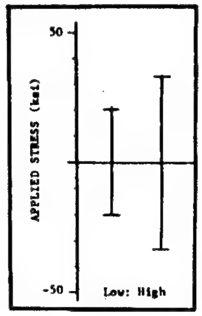
TEST S3



C.A. LIFE BLOCK 1	6.75	K-C
C.A. LIFE BLOCK 2	931	K-C
BLOCK 1 CYCLES	3	K-C

	PREDICTION	OBSERVED
DAMAGE STATE (END OF BLOCK 1)	5.6	5.2
STIFFNESS (%) (END OF BLOCK 1)	96.3	90
REMAINING LIFE		
DAMAGE (K-C)	785	619
STIFFNESS (K-C)	686	619

TEST S4



C.A. LIFE BLOCK 1	931	K-C
C.A. LIFE BLOCK 2	6.75K	K-C
BLOCK 1 CYCLES	350	K-C

	PREDICTION	OBSERVED
DAMAGE STATE (END OF BLOCK 1)	6.3	6.0
STIFFNESS (%) (END OF BLOCK 1)	95.2	96.0
REMAINING LIFE		
DAMAGE (K-C)	2.35	5.56
STIFFNESS (K-C)	2.83	5.56

CONCLUSIONS TO DATE

- Damage Accumulation is Systematic and Progressive
- Stiffness Change is An Indicator of Damage
- Delamination is the Dominant Damage Mode
- Stress Level Affects Damage Rates
- Stress Ratio Affects the Damage Accumulation Processes
- Stress Range Contributes to Damage Growth
- Tensile Stress Creates Transverse Cracks
- Compressive Stress in the Presence of Delamination Can Create Local Instabilities Which Lead to Buckling-Type Failures
- The Predictions Correlate Reasonably Well with Two-Stage Fatigue Results
- The Life Predictions are Most Accurate When the Observed Damage State and Stiffness Retention Agree with the Constant Amplitude Results.

DAMAGE ACCUMULATION AND RESIDUAL PROPERTIES OF COMPOSITES

I.M. Daniel
A. Charewicz

Illinois Institute of Technology
Chicago, IL 60616

ABSTRACT

Damage mechanisms and accumulation, and associated stiffness and residual strength reductions were studied in cross-ply graphite/epoxy laminates, under quasi-static tensile loading and tension-tension fatigue loading.

Under quasi-static loading the predominant form of damage in all specimens was transverse matrix cracking. The initiation and growth of damage is strongly dependent on laminate layup and stacking sequence. Damage was quantified in terms of a normalized ratio of layer thickness and average crack spacing. Damage is initiated at a lower stress (and strain) in the specimens with the larger number of 90-deg. plies, and it reaches the CDS at a stress level of less than 50% of the ultimate. Measured stiffness was correlated with transverse cracking. Laminate stiffness loss due to transverse matrix cracking was found to be higher than predicted. Ultimate failure was governed by the ultimate strain of the 0-deg.plies. Some variations were noted in the ultimate strain of the various laminates. These are attributed to several factors,e.g., higher scatter in the unsupported 0-deg. plies, strain concentrations at the ends of transverse cracks, and total volume of 0-deg. plies.

Stress-life data were fitted by straight lines on a log-log scale. The fatigue sensitivity decreases with the number of contiguous 90-deg. plies. Laminates, such as the $[0/90_2]_s$, have lower static strength but longer fatigue life than their stacking sequence variation $[90_2/0]_s$. Five different damage mechanisms were observed, transverse matrix cracking, dispersed longitudinal cracking, localized longitudinal cracking, delaminations along transverse cracks, and local delamination at the intersection of longitudinal and transverse cracks. The first type of damage is a low level type and develops early in the fatigue life. It reaches the CDS level long before laminate failure. However, the CDS level is not always attained if the cyclic stress is too low. Longitudinal cracking, dispersed or localized, is a particularly active form of damage. The higher fatigue sensitivity observed in the $[90_n/0]_s$ laminates corresponds to a larger extent of longitudinal cracking, as compared with the $[0/90_n]_s$ laminates.

The residual modulus measured in one laminate shows a sharp reduction initially, followed by a more gradual decrease up to failure. The residual strength showed a very large scatter in both strength and number of cycles. To minimize the effects of large scatter, all data for a given laminate were pooled together and a normalized residual strength, independent of stress level, was expressed as a function of normalized number of cycles. The characteristic features of residual strength are: sharp decrease initially, then a near plateau in the middle part of the fatigue life, and a rapid decrease in the last part of the fatigue life.

A cumulative damage model is proposed based on residual strength and the concept of equal damage curves. Although a reasonable prediction based on this model was demonstrated, the application of the model relies on a good definition of the residual strength curve. The latter is very difficult to obtain in view of the scatter. The characteristic plateau, if indeed it exists, poses a great problem as there is no one-to-one correspondence between residual strength and number of cycles over this plateau. Thus, residual strength alone, as it is measured now, does not seem to be a very discriminating measure of damage.

DAMAGE ACCUMULATION AND RESIDUAL STRENGTH DEGRADATION IN GRAPHITE/EPOXY COMPOSITES

DAMAGE MECHANISMS AND ACCUMULATION, AND ASSOCIATED STIFFNESS AND RESIDUAL STRENGTH REDUCTIONS WERE STUDIED IN CROSS-PLY GRAPHITE/EPOXY LAMINATES, UNDER QUASI-STATIC TENSILE AND TENSION-TENSION FATIGUE LOADING.

FAILURE MECHANISMS

- MATRIX CRACKING (INTRALAMINAR)
- DELAMINATION (INTERLAMINAR)
- FIBER FRACTURE AND SPLITTING
- FIBER MICROBUCKLING

METHODS OF ANALYSIS

- LAMINA-BASED ANALYSIS
- DISPERSED DEFECTS-FRACTURE MECHANICS
- PHENOMENOLOGICAL OR MACROMECHANICAL ANALYSIS

ALL OF THE ABOVE CAN BE CONDUCTED IN A DETERMINISTIC MANNER, OR PROBABILISTIC MANNER BY INTRODUCING STATISTICAL VARIATION OF DEFECTS AND STRENGTHS.

SCOPE

Material:	AS-4/3501-6 Graphite/Epoxy
Layups:	$[0/90_2]_s$
	$[90_2/0]_s$
	$[0/90_4]_s$
	$[90_4/0]_s$
Loading:	Quasi-static tension
	Tension-tension fatigue
Measurements:	Damage (X-radiography) Strain Fatigue life Residual modulus Residual strength

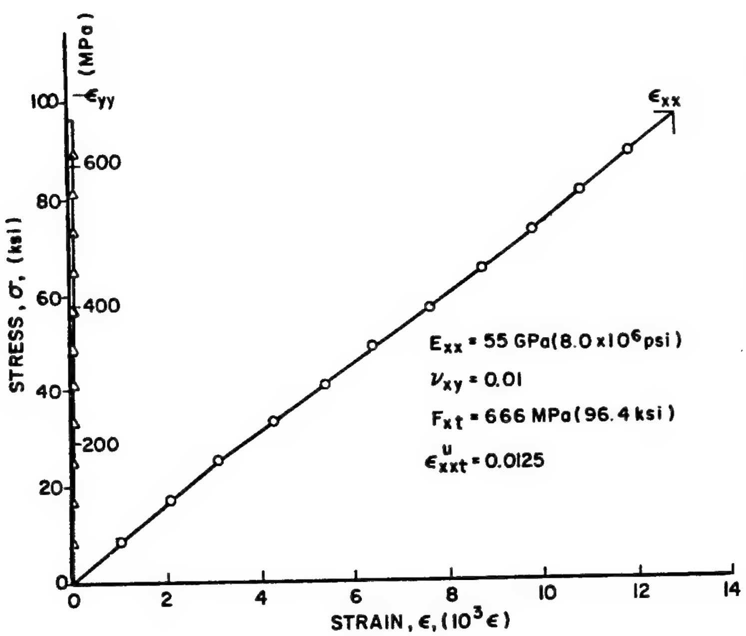


Fig. 7. Stress-Strain Curves for $[0/90_2]_s$ Graphite/Epoxy Specimen under Uniaxial Tensile Loading.

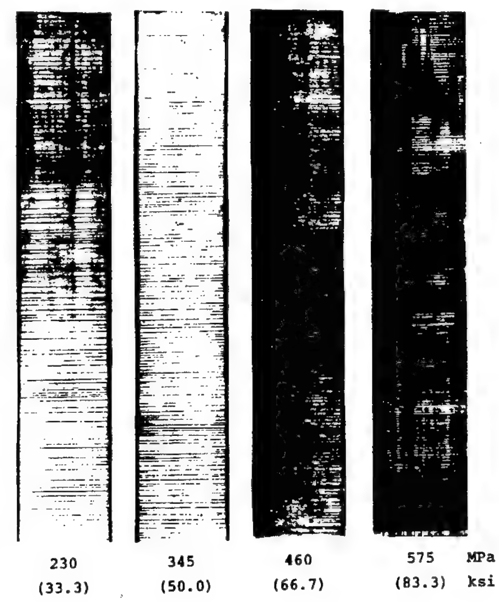


Fig. 8. X-Radiographs of a $[0/90_2]_s$ Graphite/Epoxy Laminate Under Uniaxial Tensile Loading at Various Applied Stress Levels.

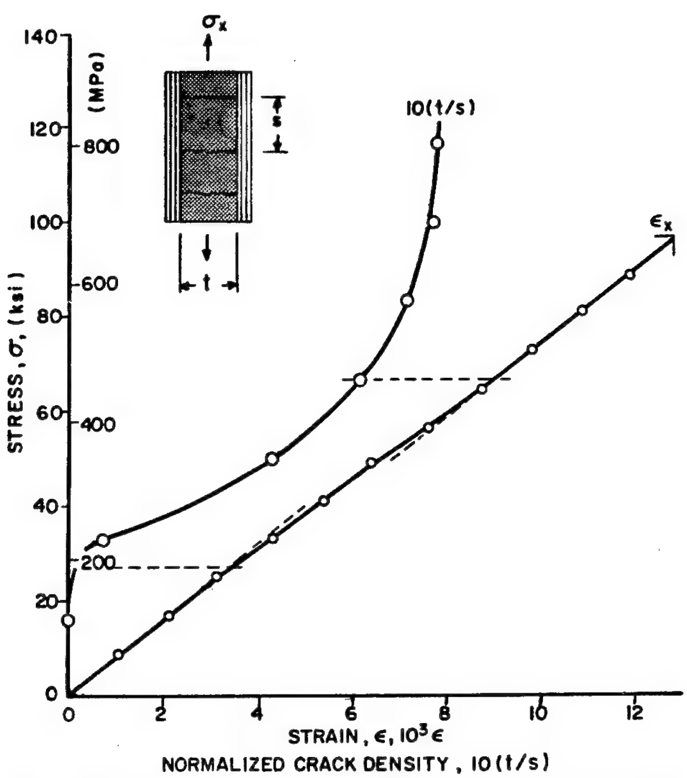


Fig. 10. Stress-Strain and Stress-vs. Crack Density Curves in $[0/90_2]_s$ Graphite/Epoxy Specimens under Uniaxial Tensile Loading.

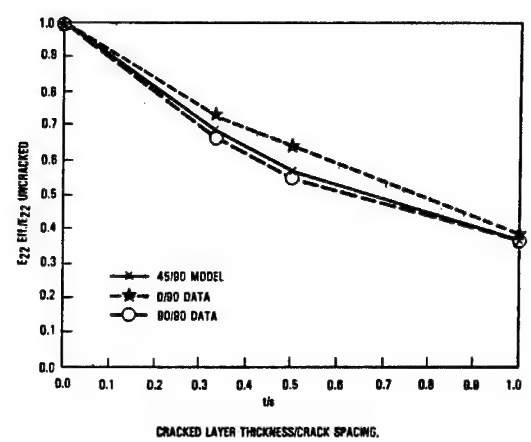


Fig. 11. Normalized Effective Transverse Modulus vs. Crack Density (Ref. 7).

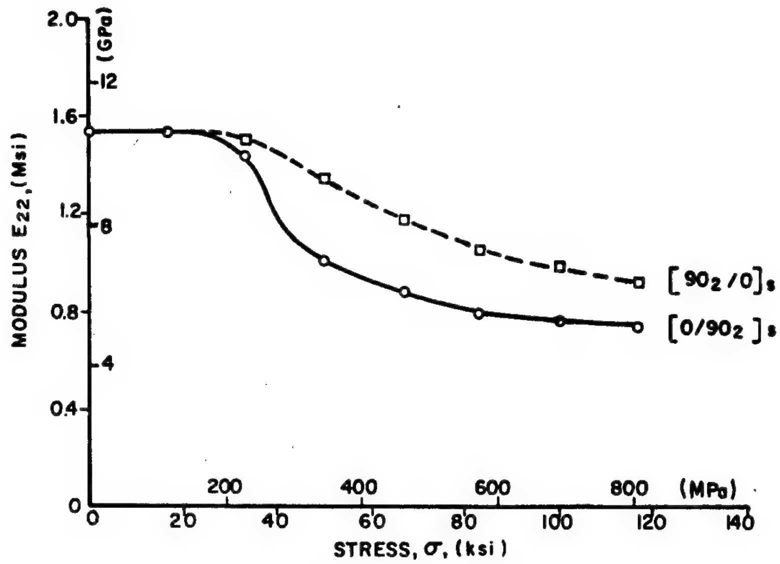


Fig. 12. Effective Transverse Modulus of 90-deg Layer in $[0/90_2]_s$ and $[90_2/0]_s$ Graphite/Epoxy Laminates under Uniaxial Tensile Loading.

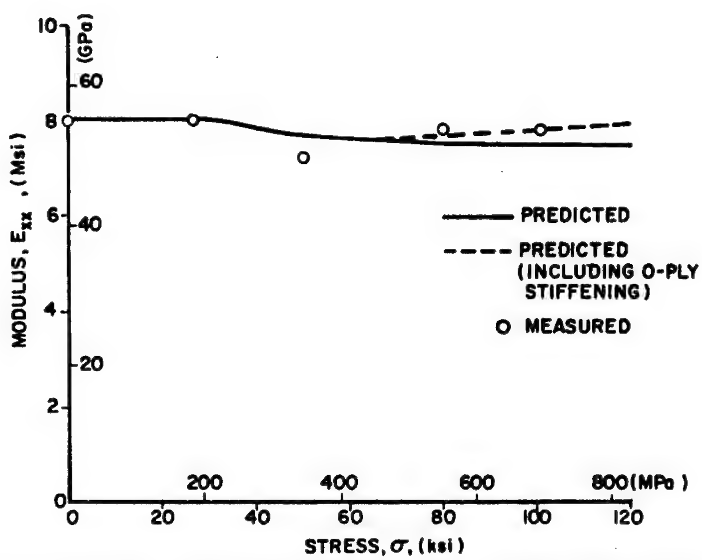


Fig. 13. Variation of Modulus with Stress for $[0/90_2]_s$ Graphite/Epoxy Laminate under Uniaxial Tensile Loading.

TABLE 1. QUASI-STATIC PROPERTIES OF AS-4/3501-6 LAMINATES UNDER UNIAXIAL TENSILE LOADING

Laminate	Width mm (in.)	Thickness mm (in.)	Modulus, E_{xx} GPa (10^6 psi)	Poisson's Ratio, ν_{xy}	Strength MPa (ksi)	Ultimate Strain ϵ^u_{xt}
$[0_6]$	12.7 (0.50)	0.762 (0.030)	145 (21.0)	0.27	2090 (303)	0.0137
$[90_{16}]$	25.4 (1.00)	2.032 (0.080)	10.6 (1.54)	0.02	64 (9.3)	0.0060
$[0/90_2]s$	25.3 (0.996)	0.762 (0.030)	54.8 (7.95)	0.01	682 (99)	0.0127
$[90_2/0]s$	25.2 (0.993)	0.787 (0.031)	56.9 (8.25)	0.02	815 (118)	0.0155
$[0/90_4]s$	25.3 (0.995)	1.270 (0.050)	32.1 (4.65)	0.01	426 (61.8)	0.0137
$[90_4/0]s$	25.4 (1.000)	1.270 (0.050)	31.8 (4.61)	0.02	435 (63.1)	0.0155

STRESS-LIFE CURVES

All data above were fitted by straight lines on a log-log scale as follows:

$$\log S = \log F_e - \frac{1}{m} \log N$$

where
S = applied cyclic stress amplitude
F_e = equivalent static strength
N = number of cycles to failure
m = constant

TABLE 2. S-N CURVE PARAMETERS

Layup	Equivalent Static Strength F_e, MPa (ksi)	Exponent, m
$[90_{16}]$	64 (9.28)	57.67
$[0/90_2]s$	779 (113)	25.42
$[90_2/0]s$	822 (119)	22.99
$[0/90_4]s$	456 (66)	50.25
$[90_4/0]s$	427 (62)	31.0

FATIGUE DAMAGE MECHANISMS

1. Transverse matrix cracking in the 90-deg. plies,
2. Dispersed longitudinal cracking in the 0-deg. plies, randomly distributed throughout the specimen,
3. Localized longitudinal cracking,
4. Delamination in the 0/90-deg. interface along transverse cracks, and
5. Local delamination at the intersection of longitudinal and transverse cracks.

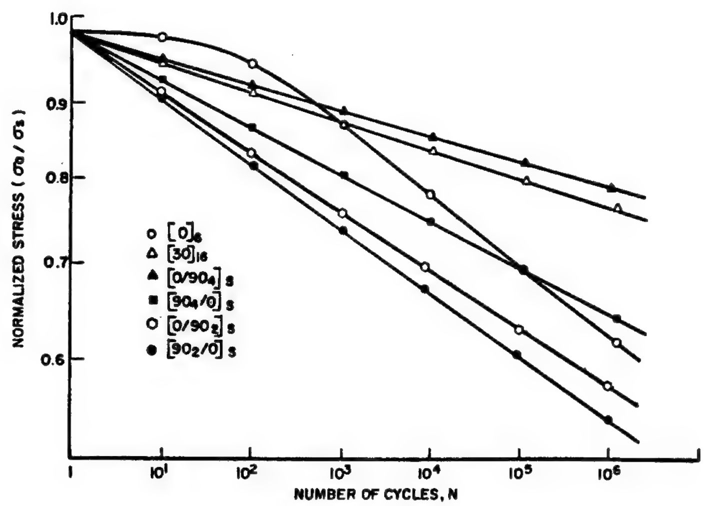


Fig. 8. Stress-Life (S-N) Curves for Unidirectional and Crossply Laminates

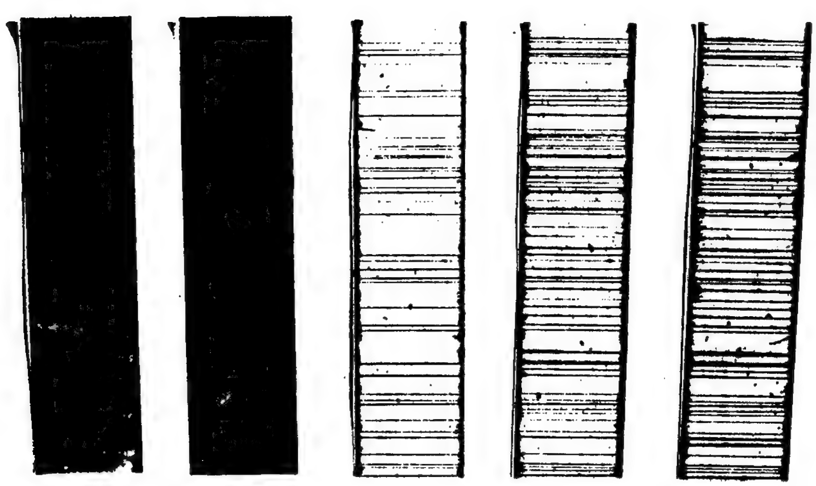


Figure 45. Transverse Cracking Growth in a $[90_2/0]_s$ laminate Tested in Fatigue at a Stress Level of 30 ksi.

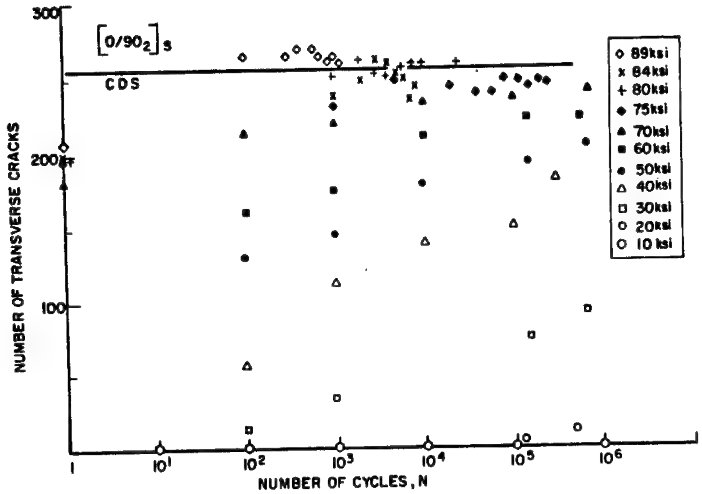


Fig. 10. Number of Transverse Cracks over a 14 cm (5.5 in.) Length in $[0/90_2]_s$ Graphite/Epoxy Laminate as a Function of Number of Cycles and Applied Stress.

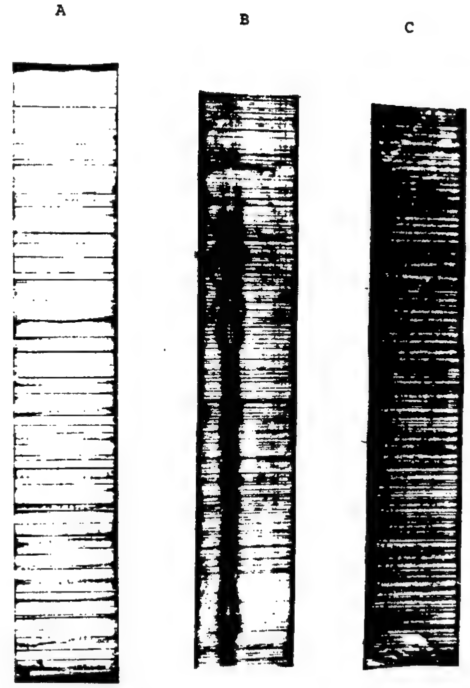
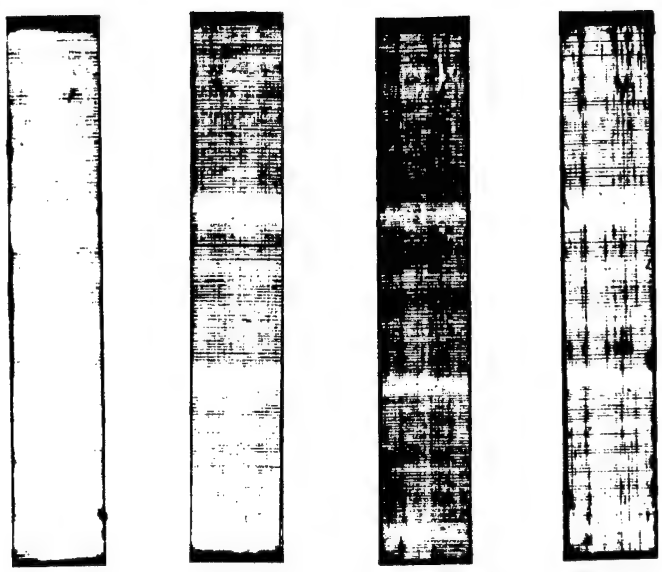


Figure 50. Different Forms of Damage Observed: (a) Dispersed Longitudinal Cracking and Transverse Cracking Delaminations; (b) Localized Longitudinal Cracking; and (c) Local Delamination at the Junction of Longitudinal and Transverse Cracks



X-Radiographs of $[0/90_2]_s$ Graphite/Epoxy Specimen Tested under Fatigue Loading with R = 0.1, σ_{max} = 518 MPa(75 ksi), for Various Numbers of Cycles.

(a) (b) (c) (d) (e)

Fig. . Failure Patterns of $[0/90_2]_s$ Graphite/Epoxy Specimens Tested under Static and Fatigue Loading Conditions.

(a) Static Test, σ_u = 616 MPa(89 ksi)

(b) Fatigue Test, R = 0.1, σ_{max} = 606 MPa(88 ksi), N=370 cycles

(c) Fatigue Test, R = 0.1, σ_{max} = 548 MPa(79 ksi), N=2,140 cycles

(d) Fatigue Test, R = 0.1, σ_{max} = 531 MPa(77 ksi), N=12,357

(e) Fatigue Test, R = 0.1, σ_{max} = 495 MPa(72 ksi), N=349,529

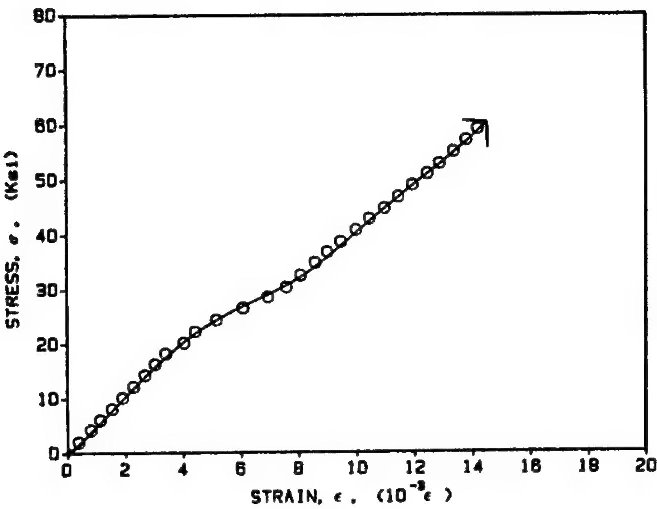
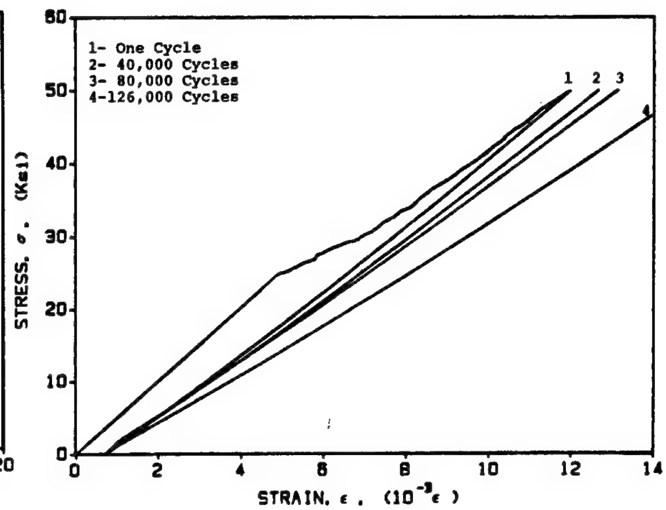
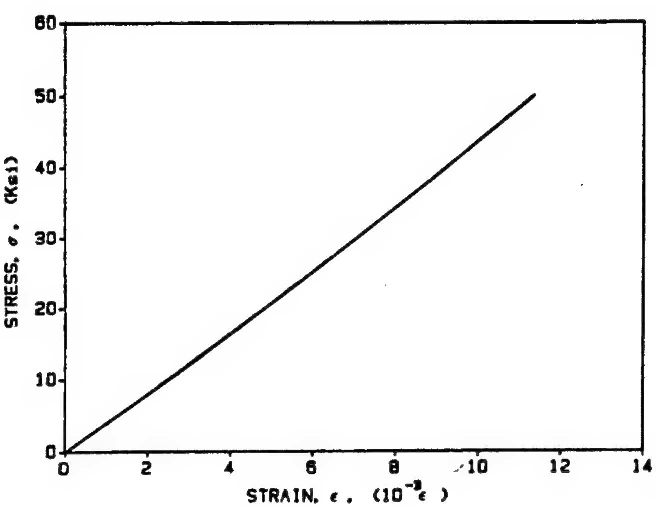


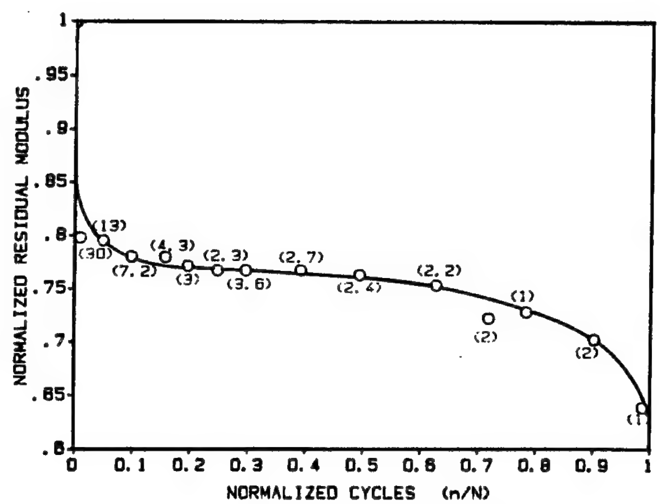
Fig. Stress-Strain Curve for $[0/90_4]_s$ Graphite/Epoxy Specimen under Uniaxial Tensile Loading.



Stress-Strain Curves for $[0/90_4]_s$ Graphite/Epoxy Specimen at Various Stages of Fatigue Life (R = 0.1)



Stress-Strain Curve for $[0/90_4]_s$ Graphite/Epoxy Specimen at Tenth Cycle of Fatigue Life



Normalized Residual Modulus for $[0/90_4]_s$ Graphite/Epoxy Specimen under Fatigue Loading at R = 0.1 and σ_{max} = 345 MPa (50 ksi)

RESIDUAL STRENGTH

$$\frac{F_r - \sigma_a}{F_o - \sigma_a} = g\left(\frac{n}{N}\right)$$

where, F_r = residual strength after n cycles
F_o = static strength
σ_a = applied cyclic stress
N = number of cycles to failure at stress σ_a

$g(\frac{n}{N})$ = function of the normalized number of cycles which must satisfy the following conditions:

$g(0) = 1$

$g(1) = 0$

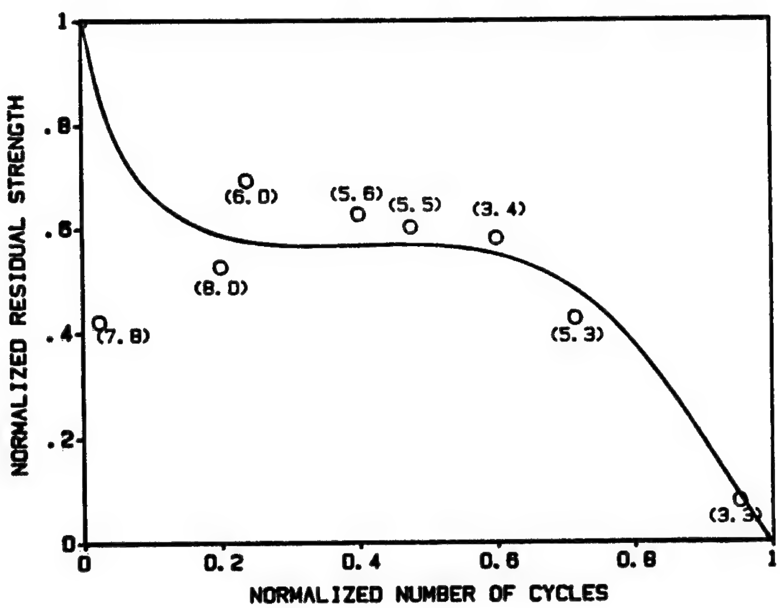
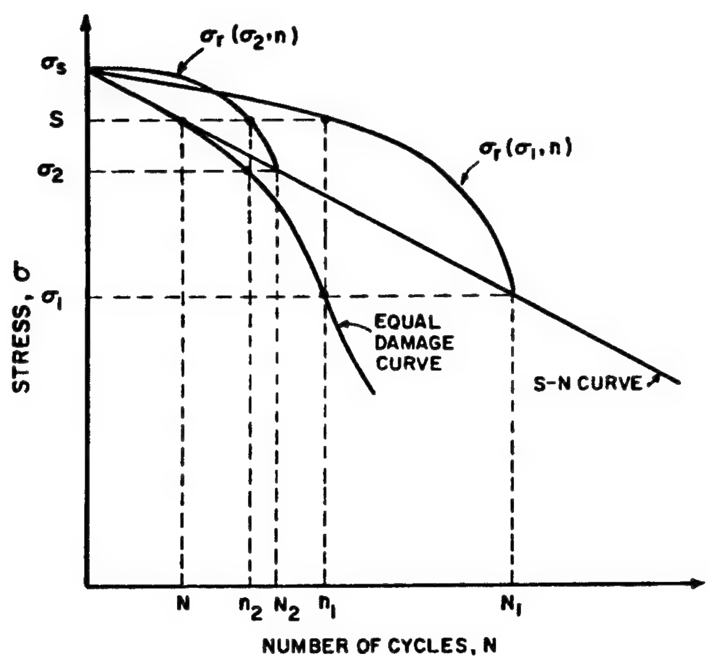


Fig. 19. Normalized Residual Strength vs. Normalized Number of Cycles for $[0/90_2]_s$ Graphite/Epoxy Laminate (Numbers in parentheses refer to number of specimens weighted).



Prediction of Residual Life for Two-Step Loading by Means of Equal Damage Curve.

CUMULATIVE DAMAGE MODEL

Normalized change in residual strength:

$$\frac{1 - f_r}{1 - s} = D(\frac{n}{N}, s)$$

where $f_r = \frac{F_r}{F_o}$ = normalized residual strength

F_r = residual strength

F_o = static strength

$s = \frac{\sigma}{F_o}$ = normalized applied cyclic stress

D = damage function

N = number of cycles to failure at stress s

This damage function must satisfy the following conditions:

$D(0,s) = 0$ for all s

$D(1,s) = 1$ for all s

Since the applied stress s is related to the number of cycles to failure N by the S-N curve

$$s = f(N)$$

the damage function can be expressed as

$$D[\frac{n}{N}, f(N)] = \frac{1 - f_r}{1 - s}$$

CUMULATIVE DAMAGE PREDICTION

Damage rate:

$$\frac{dD}{dn} = F(\frac{n}{N}, s, D) = F[\frac{n}{f^{-1}(s)}, s, D]$$

The number of cycles N_f required to increase the damage from an initial value D_i to a final value D_f is

$$N_f = \int_{D_i}^{D_f} \frac{dD}{F(\frac{n}{N}, s, D)}$$

In the case of spectrum fatigue loading, the damage produced after the k^{th} block of loading is expressed as

$$D_k = \sum_{i=1}^{k} \int_0^{n_i} F(\frac{n_i}{N}, s_i, D_i)\,dn$$

CONCLUSIONS

QUASI-STATIC LOADING

1. THE $[0/90_2]_s$ AND $[90_2/0]_s$ LAMINATES EXHIBIT A STRESS-STRAIN BEHAVIOR CONSISTING OF THREE LINEAR RANGES, AN INITIAL ONE PREDICTED BY LAMINATION THEORY, A SECOND RANGE OF REDUCED STIFFNESS, AND A THIRD ONE OF INCREASED STIFFNESS.
2. THE $[0/90_4]_s$ LAMINATE EXHIBITED ONLY TWO LINEAR RANGES.
3. ULTIMATE FAILURE IN ALL LAMINATES WAS GOVERNED BY THE ULTIMATE STRAIN IN THE 0-DEG PLIES, WITH VARIATIONS ATTRIBUTED TO HIGHER SCATTER IN UNSUPPORTED 0-DEG PLIES, STRAIN CONCENTRATIONS AT THE ENDS OF TRANSVERSE CRACKS, AND VOLUME OF 0-DEG PLIES.
4. TRANSVERSE CRACKING WAS THE PREDOMINANT FORM OF DAMAGE IN ALL LAMINATES, WITH DAMAGE INITIATION AND GROWTH STRONGLY DEPENDENT ON LAMINATE LAYUP.
5. DAMAGE IS INITIATED AT A LOWER STRESS (AND STRAIN) IN SPECIMENS WITH THE LARGER NUMBER OF 90-DEG PLIES, AND IT REACHES THE CHARACTERISTIC DAMAGE STATE (CDS) AT A STRESS OF LESS THAN 50% OF THE ULTIMATE.
6. THE LIMITING CRACK DENSITY WAS HIGHEST FOR THE $[0/90_2]_s$ LAMINATE AND LOWEST FOR THE $[90_4/0]_s$ LAMINATE.
7. AT FAILURE ALL SPECIMENS SHOWED LONGITUDINAL CRACKING IN THE 0-DEG PLIES, ACCOMPANIED BY LOCALIZED DELAMINATIONS.
8. LAMINATE STIFFNESS LOSS DUE TO TRANSVERSE CRACKING WAS FOUND TO BE HIGHER THAN PREDICTED.

CONCLUSIONS

CYCLIC LOADING

1. STRESS-LIFE DATA WERE FITTED BY STRAIGHT LINES ON A LOG-LOG SCALE. THE FATIGUE SENSITIVITY DECREASES WITH THE NUMBER OF CONTIGUOUS 90-DEG PLIES.
2. LAMINATES OF $[0/90_2]_s$ AND $[0/90_4]_s$ LAYUP HAD LOWER STATIC STRENGTH BUT LONGER FATIGUE LIFE THAN THEIR CORRESPONDING STACKING SEQUENCE VARIATIONS $[90_2/0]_s$ AND $[90_4/0]_s$.
3. FIVE DIFFERENT DAMAGE MECHANISMS WERE OBSERVED: TRANSVERSE MATRIX CRACKING, DISPERSED LONGITUDINAL CRACKING, LOCALIZED LONGITUDINAL CRACKING, DELAMINATION ALONG TRANSVERSE CRACKS, AND LOCAL DELAMINATION AT INTERSECTIONS OF TRANSVERSE AND LONGITUDINAL CRACKS.
4. THE INITIAL DAMAGE MECHANISM IS TRANSVERSE MATRIX CRACKING, WHICH REACHES THE CDS LEVEL IN MOST CASES AT FEWER THAN 1,000 CYCLES.
5. LONGITUDINAL CRACKING, DISPERSED OR LOCALIZED, IS AN ACTIVE FORM OF DAMAGE AND IS A PRECURSOR TO FAILURE.
6. THE HIGHER FATIGUE SENSITIVITY OF THE $[90_n/0]_s$ LAMINATES CORRESPONDS TO A LARGER EXTENT OF LONGITUDINAL CRACKING, COMPARED TO THE $[0/90_n]_s$ LAMINATES.
7. THE RESIDUAL MODULUS SHOWS A SHARP REDUCTION INITIALLY, FOLLOWED BY A MORE GRADUAL DECREASE UP TO FAILURE.
8. RESIDUAL STRENGTH SHOWS A SHARP DECREASE INITIALLY, THEN A PLATEAU IN THE MIDDLE PART OF THE FATIGUE LIFE, AND A RAPID DECREASE IN THE LAST PART OF THE FATIGUE LIFE.
9. A TENTATIVE CUMULATIVE DAMAGE MODEL IS PROPOSED BASED ON RESIDUAL STRENGTH, WITH REASONABLE PREDICTIVE CAPABILITY. HOWEVER, RESIDUAL STRENGTH ALONE WITH ITS LARGE SCATTER DOES NOT SEEM TO BE A VERY DISCRIMINATING MEASURE OF DAMAGE.

CUMULATIVE DAMAGE MODEL FOR ADVANCED COMPOSITE MATERIALS

H. R. Miller
General Dynamics
Fort Worth Division

K. L. Reifsnider
Virginia Polytechnic Institute
and State University

D. A. Ulman
General Dynamics
Fort Worth Division

W. W. Stinchcomb
Virginia Polytechnic Institute
and State University

ABSTRACT

The objective of this program is to develop a cumulative damage model for advanced composite materials. The effort to achieve this objective involves the development of analytical models and experimental procedures for accurately predicting and characterizing the mechanical responses of advanced composites subjected to conditions representative of those seen in service. Throughout the development, refinement, and verification phases of the program, the investigation has focused on addressing the residual strength and residual life of symmetric composite laminates. The modeling activity has been closely linked with the generation of stiffness, strength, and damage development data obtained from unnotched coupons subjected to a variety of loading conditions.

The model as developed and refined to date has the following salient features: (1) It predicts the strength and life of engineering composite laminates under tension-tension, compression-compression, tension-compression, block spectrum loading, and constant amplitude cyclic loading; (2) it replaces Miner's rule with an engineering model based on the physical mechanisms of damage and failure; and, (3) it accounts for effects such as load sequence, biaxial stress state in critical elements, lamination differences, strength and stiffness differences, and laminate response difference. In addition, the model has been cast in a modular format that allows the integration of new or improved methodologies as they are developed.

This program is sponsored by the Air Force Wright Aeronautical Laboratories, Materials Laboratory, under Contract No. F33615-81-C-5049. Mr. Marvin Knight is the Air Force Project Engineer. Results of the development and refinement phases of the program have been documented in AFWAL-TR-82-4094 and AFWAL-TR-84-4007, respectively.

CUMULATIVE DAMAGE MODEL
FOR
ADVANCED COMPOSITE MATERIALS

(F33615-81-C-5049)

H. R. Miller
D. A. Ulman

General Dynamics
Fort Worth Division

K. L. Reifsnider
W. W. Stinchcomb

Virginia Polytechnic
Institute & State University

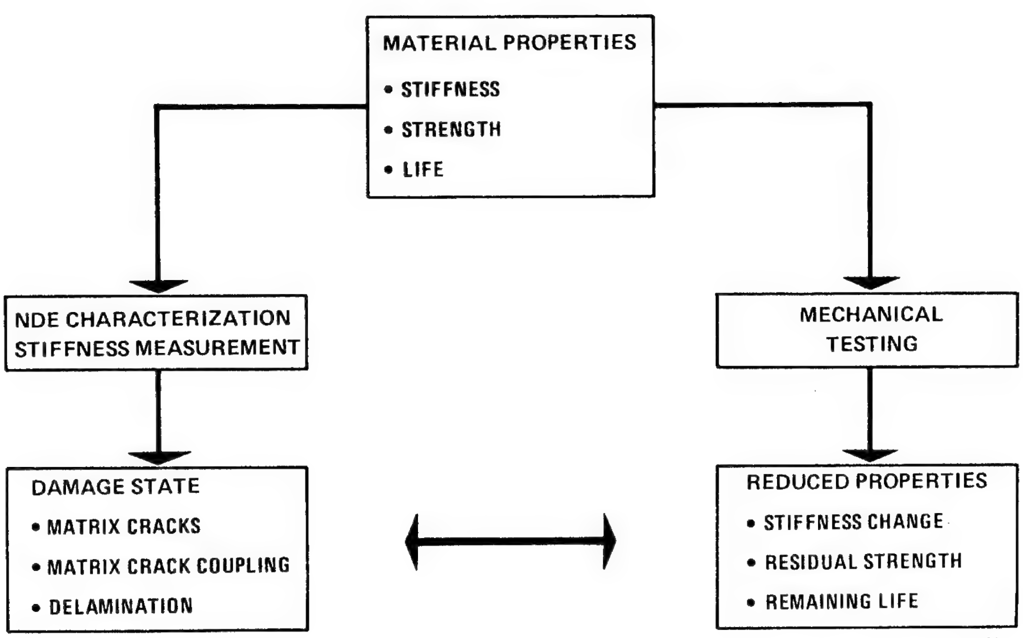
PROGRAM OBJECTIVE

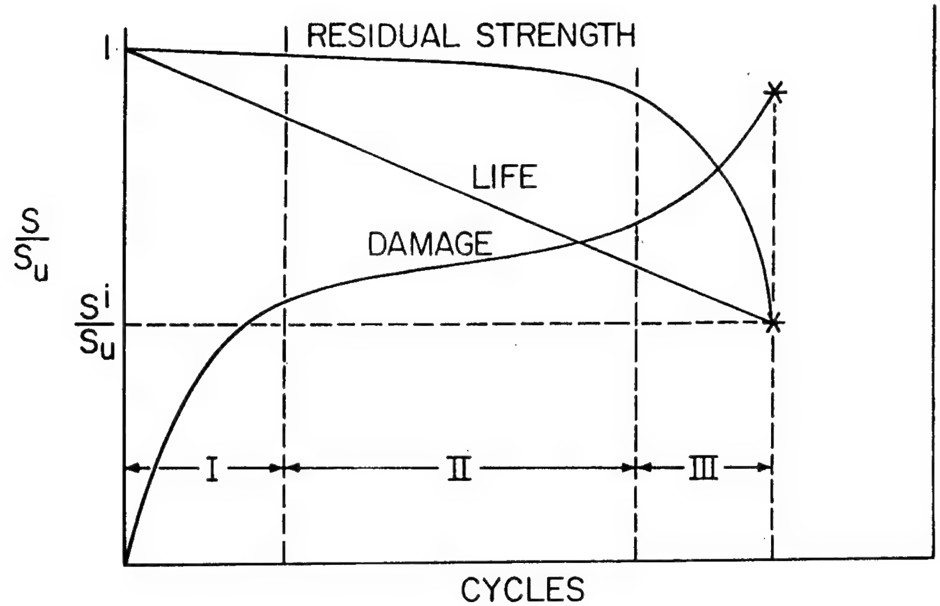
DEVELOP A CUMULATIVE DAMAGE MODEL FOR ADVANCED COMPOSITE MATERIALS CAPABLE OF PREDICTING THE STRENGTH, STIFFNESS, AND LIFE OF ARBITRARY SYMMETRIC COMPOSITE LAMINATES SUBJECTED TO A VARIETY OF COMPLEX LOAD HISTORIES.

THREE PHASE PROGRAM:
- PHASE I - MODEL DEVELOPMENT
- PHASE II - MODEL REFINEMENT
- PHASE III- MODEL VERIFICATION

CUMULATIVE DAMAGE MODEL FOR ADVANCED COMPOSITE MATERIALS

PROPERTY DEGRADATION APPROACH



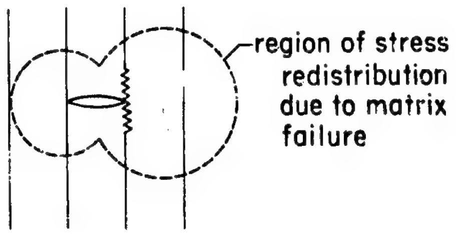


SCHEMATIC DIAGRAM OF REGIONS OF DAMAGE DEVELOPMENT FOR COMPOSITE LAMINATES

LAMINATES STUDIED

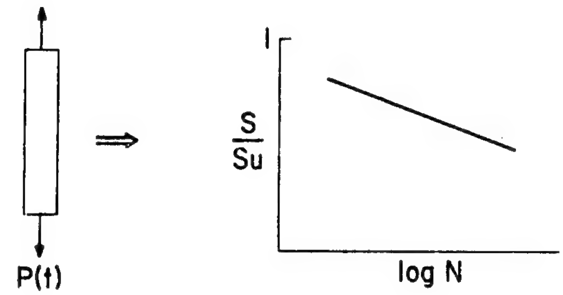
DESIGNATION	FAMILY	STACKING SEQUENCE
A	(25/50/25)	$[(0/+45/-45/90)_S]_{3S}$
B	(25/50/25)	$[(0/90/+45/-45)_S]_{3S}$
C	(25/50/25)	$[(0/+45/90/-45)_S]_{3S}$
D	(10/45/45)	$[0/((+45/-45/90)_S/90/90/+45/-45/90)_S/0]_S$
E	(40/50/10)	$[(0/+45/0/-45)_S/(0/+45/-45/90)_S/(0/+45/-45/0)_S]_S$
F	(33/67/ 0)	$[(0/+45/-45)_S]_{4S}$

Matrix: "Subcritical Elements"



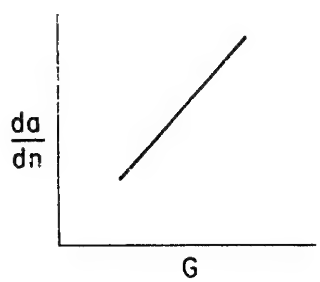
$\Rightarrow$ State of stress, rate of degradation changes in unbroken Plies.

Fibers: "Critical Elements"

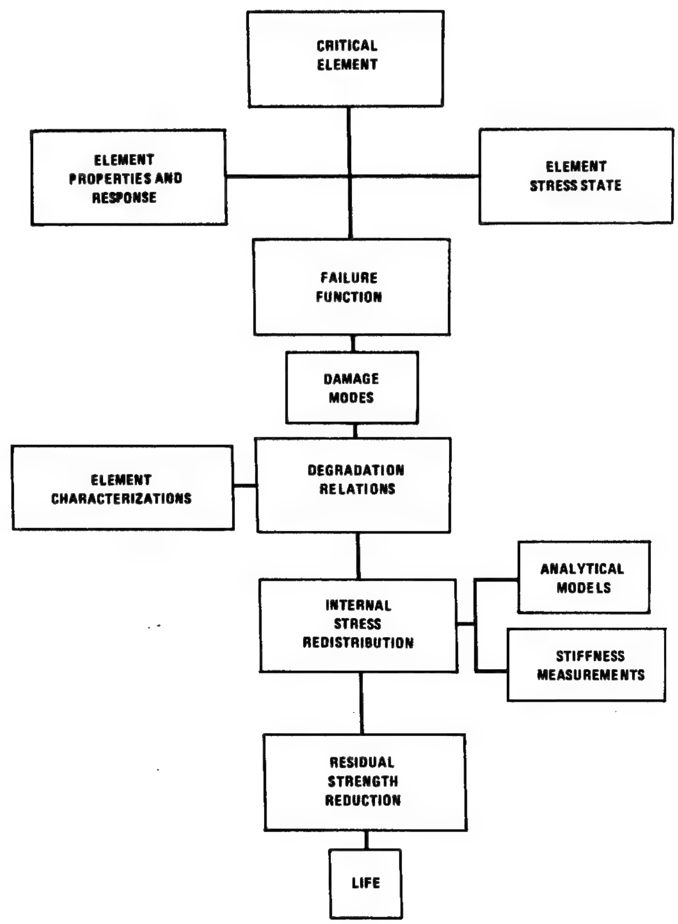


$$\Rightarrow \frac{S}{Su} = A - \frac{B}{(\log N)^x}$$

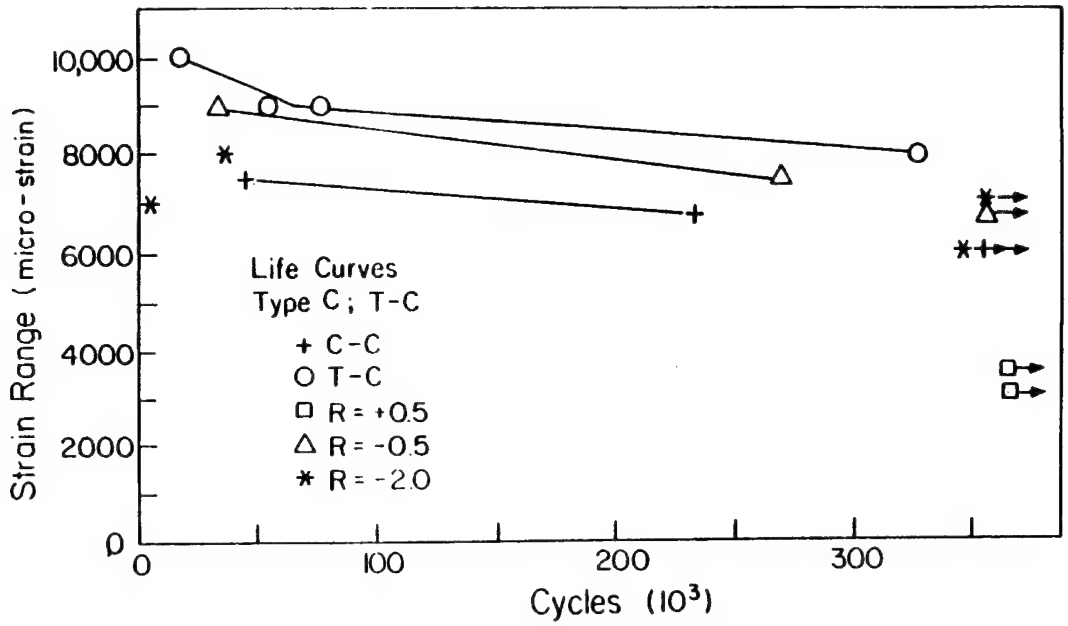
Delamination:



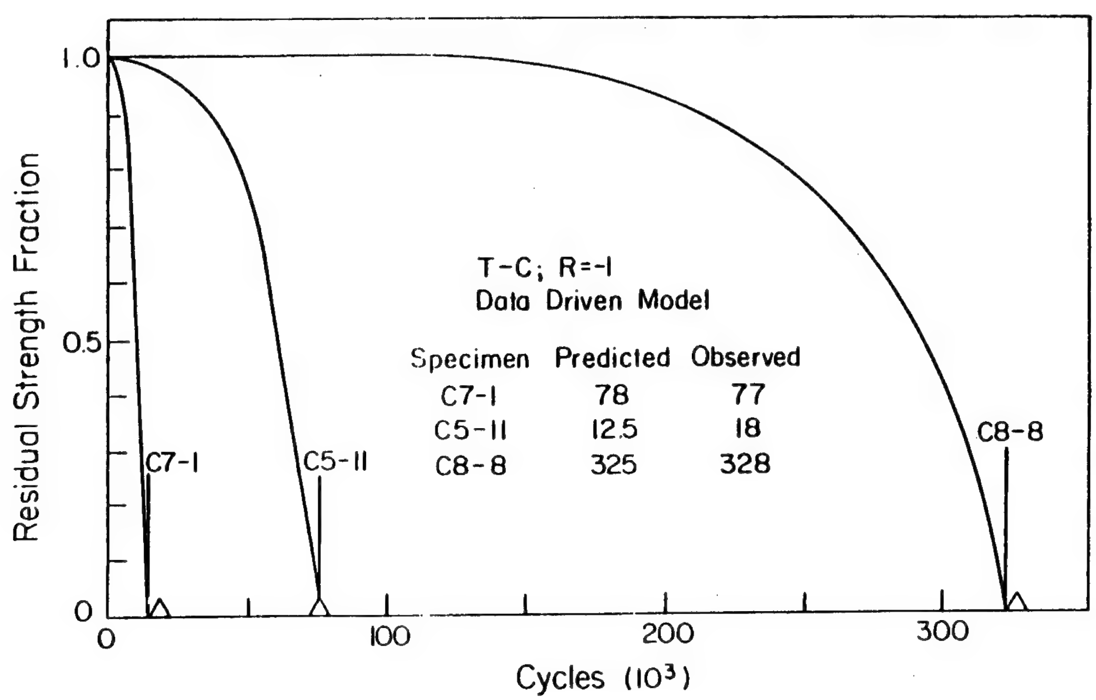
$$\Rightarrow \frac{da}{dn} = \alpha\, G^{\beta}$$



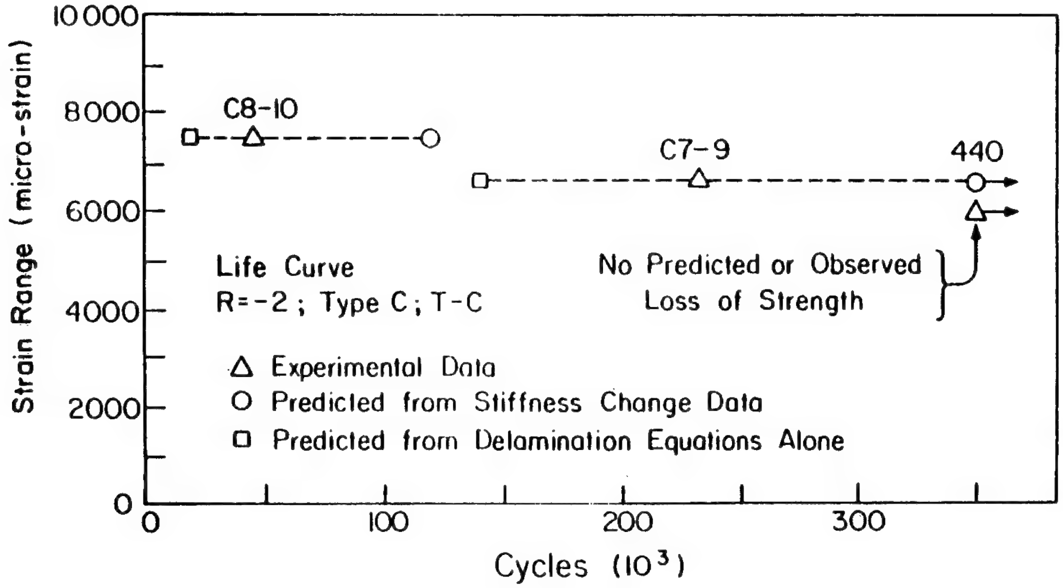
FLOW DIAGRAM OF MODEL USE PROCEDURE.



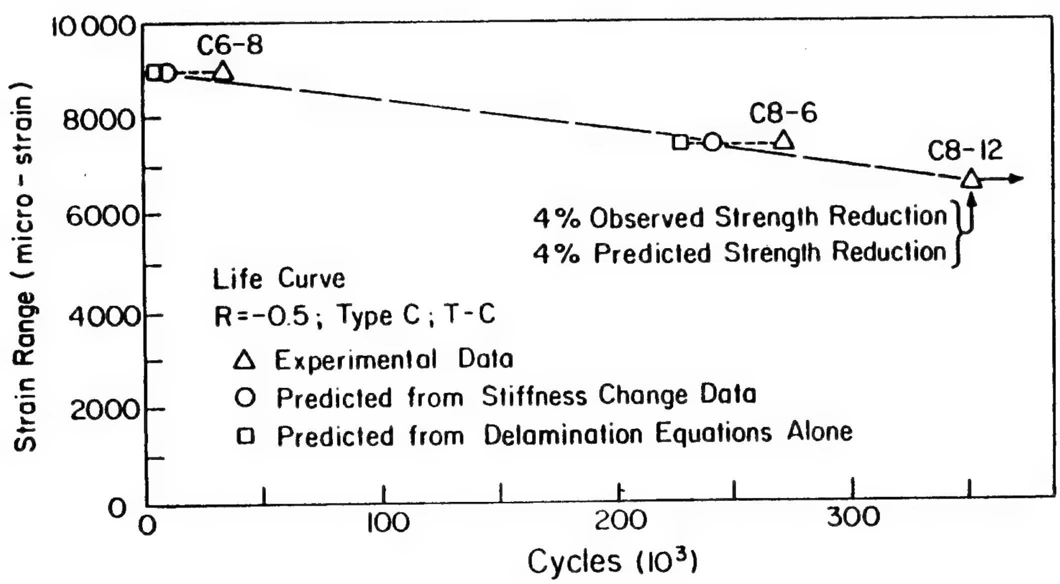
FATIGUE LIFE OBSERVATIONS FOR FIVE DIFFERENT R-VALUES



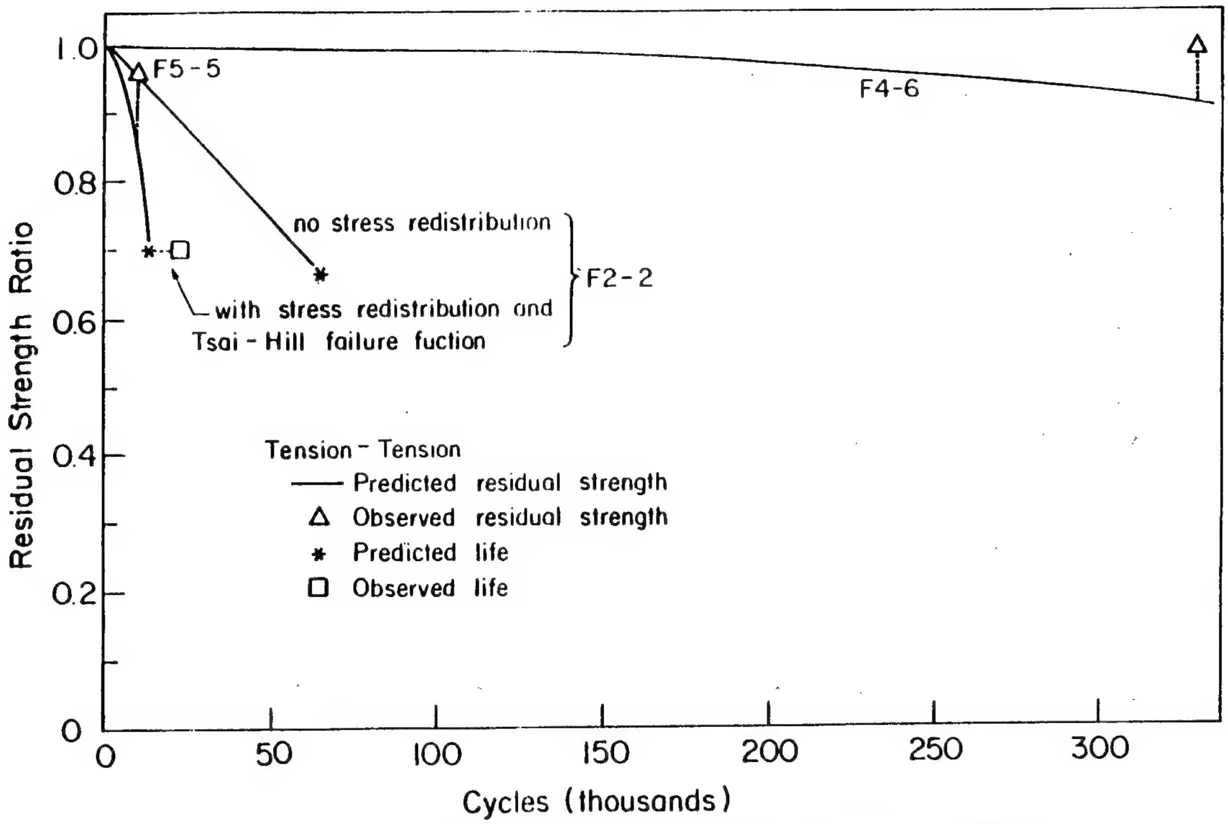
RESIDUAL STRENGTH PREDICTIONS FOR R=-1, TYPE C LAMINATE



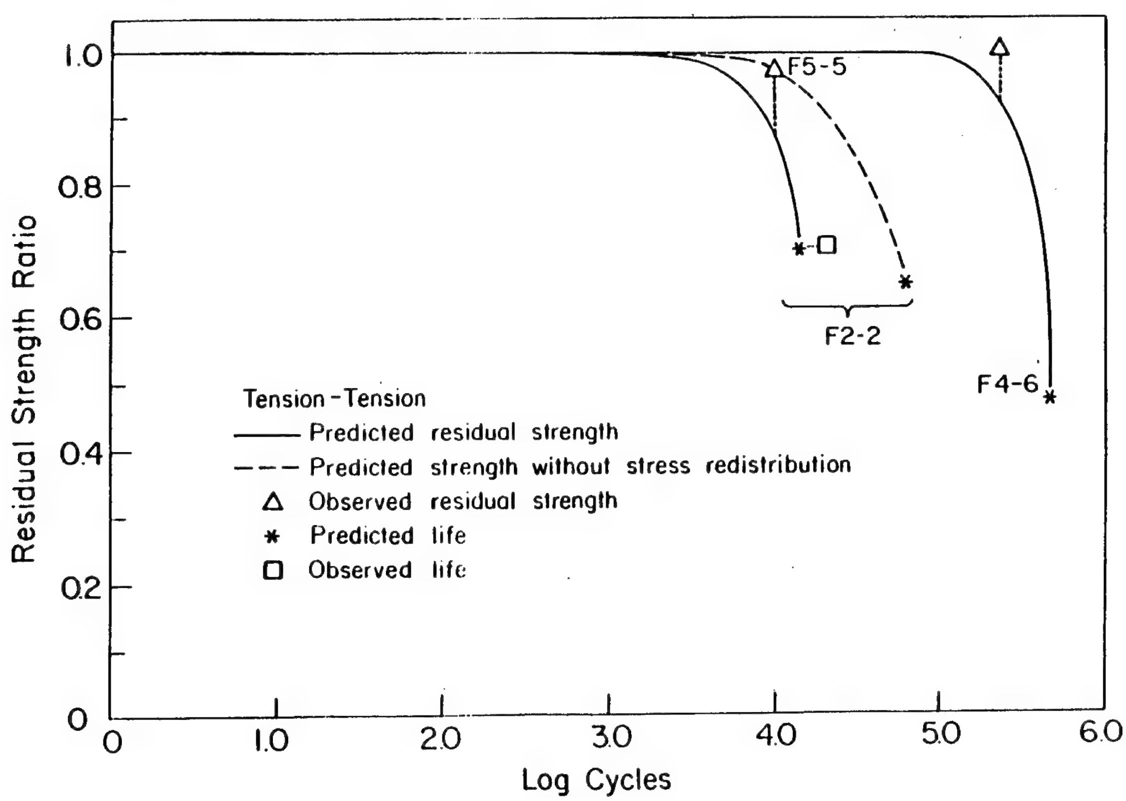
PREDICTED AND OBSERVED LIFE FOR R=-2 TESTS



PREDICTED AND OBSERVED LIFE FOR R=-0.5 TESTS



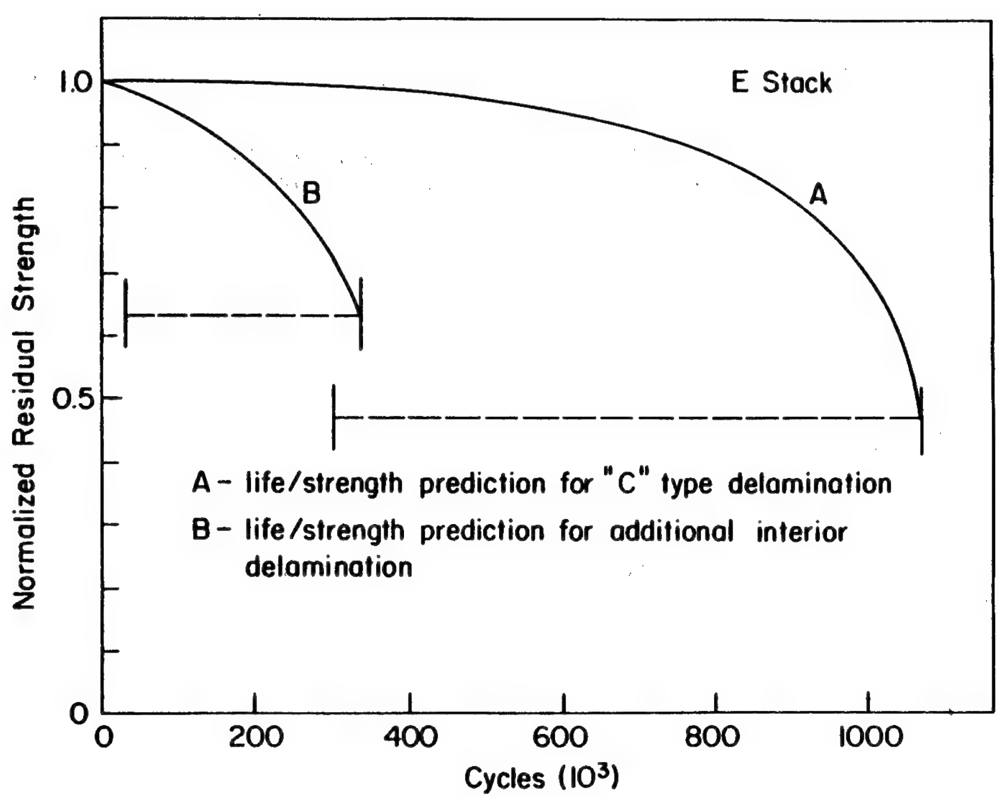
PREDICTIONS & OBSERVATIONS FOR TENSION-TENSION LOADING TYPE F LAMINATE



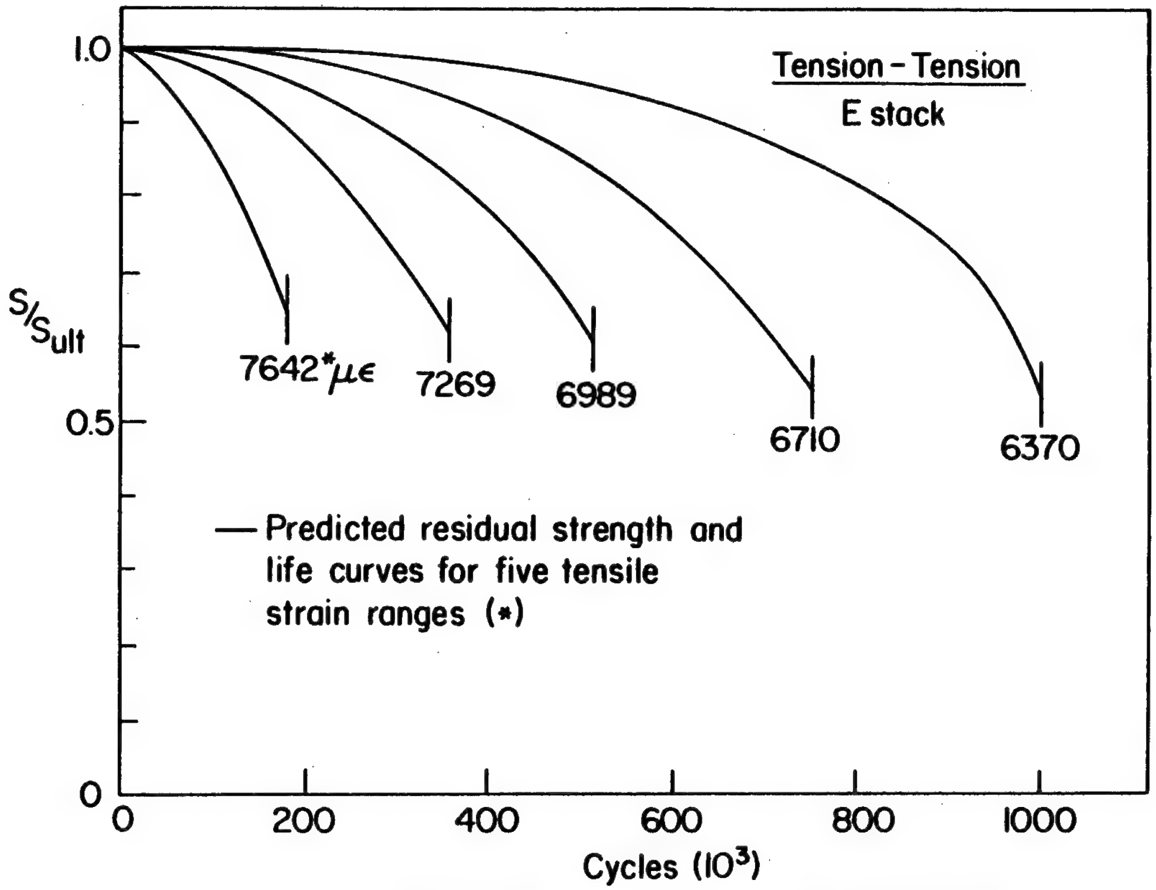
PREDICTIONS & OBSERVATIONS FOR TENSION-TENSION LOADING TYPE F LAMINATE

Phase III Test Matrix

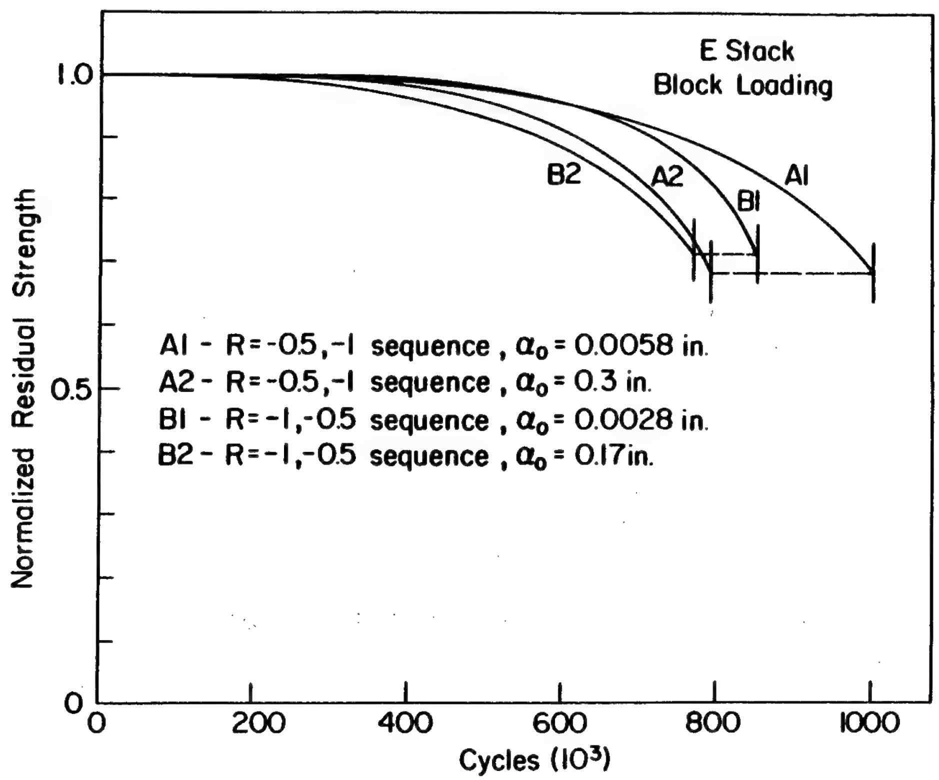
	Stacking Sequence	Width	Material	Loading Conditions	Number of Specimens
Stacking Sequence	"E"	1"	AS /3502	T-T T-C Block	9
Width	"C"	2"	AS /3502	T-C	3
Material	"C"	1"	Gr/BMI	T-C Block	6
Loading Condition	"C"	1"	AS /3502	Block 1 Block 2 Block 3	9



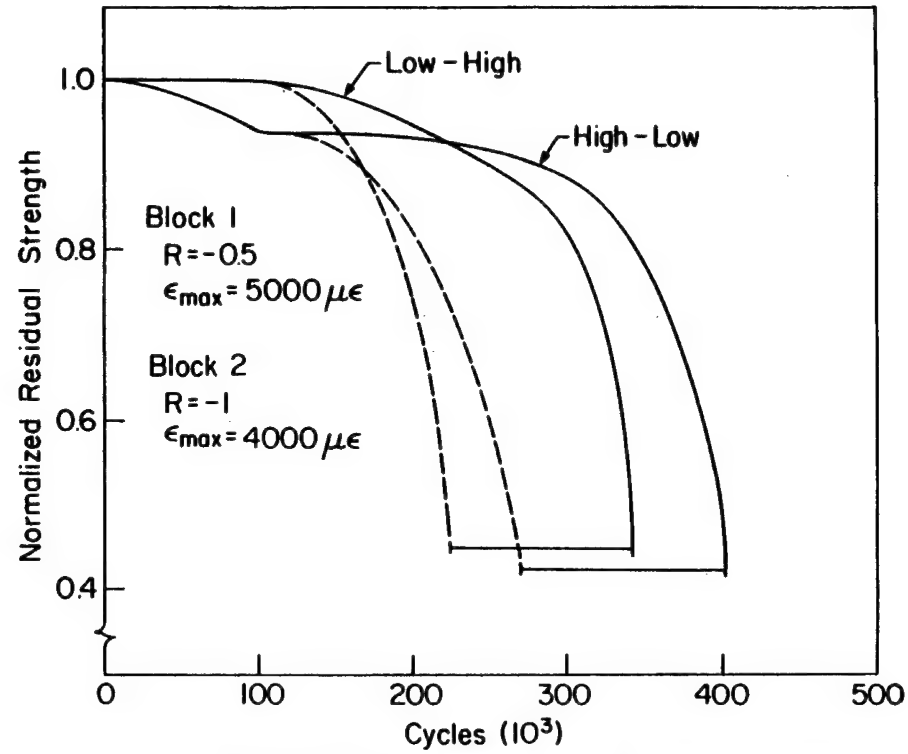
PREDICTIONS FOR TENSION-COMPRESSION LOADING TYPE E LAMINATE



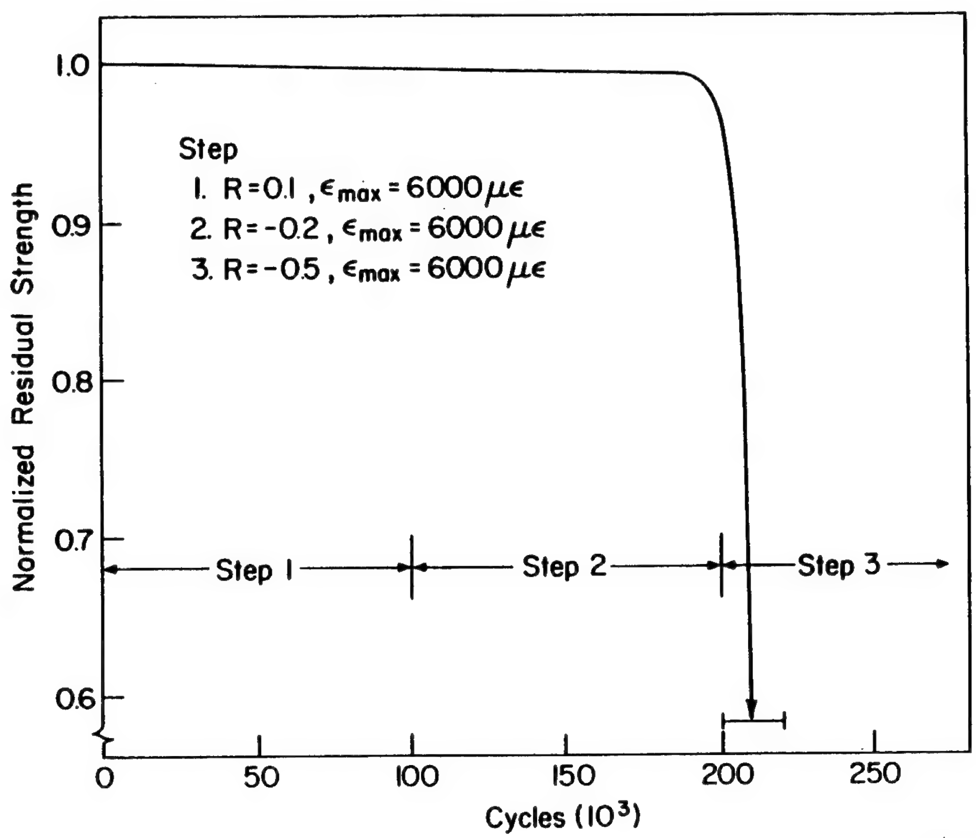
PREDICTIONS FOR TENSION-TENSION LOADING
TYPE E LAMINATE



PREDICTIONS FOR TWO BLOCK LOADING SEQUENCES
TYPE E LAMINATE



PREDICTIONS FOR REVERSED BLOCK LOADING SEQUENCE TYPE C LAMINATE



PREDICTIONS FOR A BLOCK LOADING SEQUENCE TYPE C LAMINATE

identify mat'l used in plots or presumption that is common?

SUMMARY

THE MODEL PREDICTS THE STRENGTH AND LIFE OF ENGINEERING COMPOSITE LAMINATES UNDER TENSION-TENSION, TENSION-COMPRESSION, COMPRESSION-COMPRESSION BLOCK SPECTRUM LOADING, AND UNDER CONSTANT AMPLITUDE CYCLIC LOADING.

THE MODEL ACCOUNTS FOR EFFECTS SUCH AS LOAD SEQUENCE, BIAXIAL STRESS STATE IN CRITICAL ELEMENTS, LAMINATION DIFFERENCES, AND STRENGTH AND STIFFNESS DIFFERENCES.

THE MODEL REPLACES MINER'S RULE WITH AN ENGINEERING MODEL BASED ON THE PHYSICAL MECHANISMS OF DAMAGE AND FAILURE.

1. - NDI
Damage State
2. - [illegible] on plot

[illegible]:
[illegible] - How do you handle increasing strength?
(Tom's?)
Note [illegible] - In standard fatigue is no problem
are we wasting time?
(future operating loads will be higher than static proof)

INTERLAMINAR FRACTURE OF GRAPHITE/EPOXY COMPOSITES UNDER TENSILE AND COMPRESSIVE LOADING

L. W. Rehfield, E. A. Armanios and A. D. Reddy

School of Aerospace Engineering
Georgia Institute of Technology
Atlanta, Georgia 30332

ABSTRACT

Interlaminar fracture or delamination is a primary damage mode in laminated composites. It is caused by high interlaminar stresses which are produced by local stress raisers such as holes, free edges, ply drops and other defects and discontinuities which may be manufacturing-related or service induced. Delaminations alter internal load paths and usually contribute to the ultimate failure of the structure. Interlaminar fracture toughness, which characterizes the resistance to delamination, is a key parameter in describing the damage tolerance of laminated composite materials.

The present work is concerned with the development of an interlaminar fracture specimen and test for Mode II dominated behavior and efficient means of analyzing and understanding test results. A unique feature is that the specimen is designed to be tested in both net tension and compression.

The heart of the analysis approach is a simple, new method which permits the rapidly varying interlaminar stresses to be determined by elementary means. Extensive numerical computations are thus avoided and the results are obtained in closed form. The parameters governing the behavior are easily identified. The specimen is a symmetric double lap shear type specimen with end tabs. The geometry of the notch and tabs is designed to precipitate crack growth in the gage section. The layup is selected to produce balanced symmetric specimens with differently oriented interface plies at the notch. This also minimizes the nesting effect at the interface. Overall dimensions of the specimen are determined by the boundary layer effects from tab ends and from the notch. Interaction among the local stresses in the above regions is avoided. The lengths of notched and unnotched regions are selected so as to provide an adequate crack length to be monitored.

Experiments under both net tensile and compressive forces have been conducted on double-cracked-lap-shear (CLS) specimens and single CLS specimens made of AS4/3502 graphite/epoxy material. It is observed that under static tensile loading a specimen exhibits increasing resistance to crack growth, which is a stable fracture process, until an unstable, catastrophic, ultimate fracture occurs which fails the specimen. Compressive behavior is quite different. No crack growth is observed in compression prior to a single, unstable, catastrophic fracture event which fails the specimen.

Preliminary observation of the fracture surfaces in the failed specimens indicate different characteristics for specimens tested under tensile and compressive loading. A detailed fracture surface analysis is in progress.

OBJECTIVES

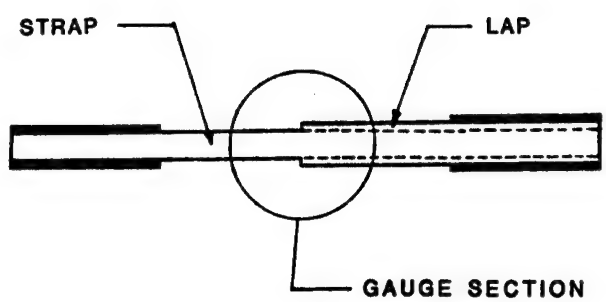
- DEVELOP MODE II INTERLAMINAR FRACTURE SPECIMEN AND TEST

- DEVELOP EFFICIENT MEANS OF ANALYZING AND UNDERSTANDING TEST RESULTS

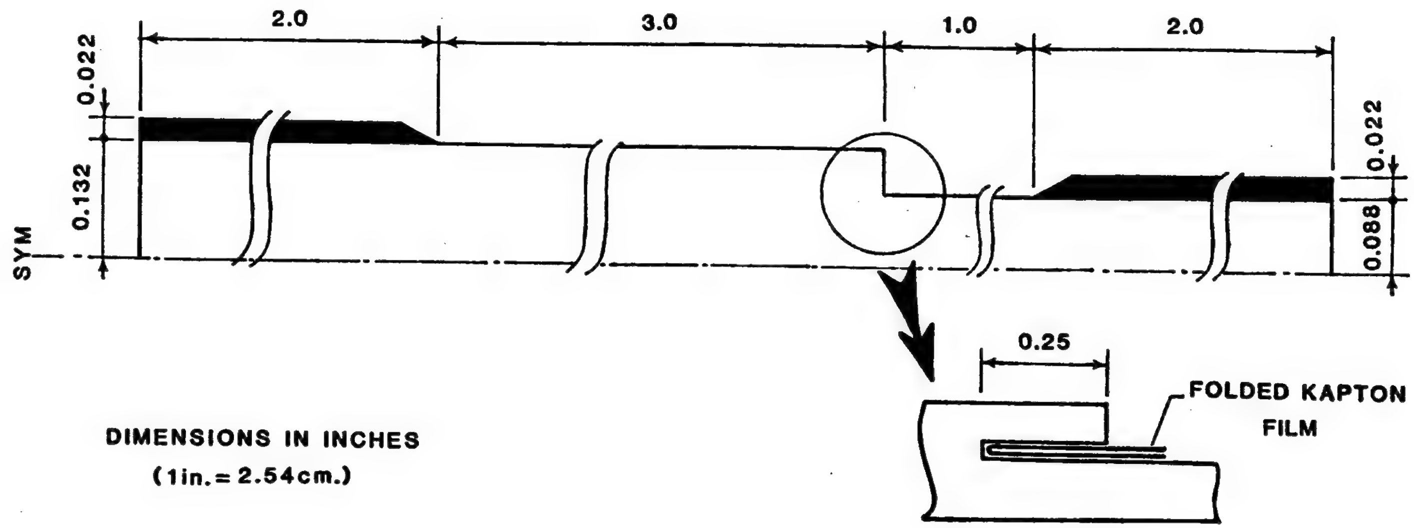
- **TENSION AND COMPRESSION**

SCIENTIFIC APPROACH

- DEVELOP SUBLAMINATE ANALYSIS
  - VALIDATION STUDY
  - DESIGN
  - INTERPRET TEST DATA
- DESIGN AND MANUFACTURE SPECIMENS
- PERFORM TENSION AND COMPRESSION EXPERIMENTS
- OBSERVE AND ANALYZE RESULTS



DOUBLE-LAP FRACTURE SPECIMEN



SPECIMEN DESIGN

TENSION/COMPRESSION TESTING

TENSION

- CRACK GROWTH IS INTERMITTENT
- P_c DEPENDS ON CRACK LENGTH
- G IN PROPER RANGE
- SPECIMENS BEHAVE AS DESIGNED
  - NO CRACK WANDERING

COMPRESSION

- NO STABLE CRACK GROWTH
- SUDDEN FAILURE
  - FAILURE CRITERIA APPEARS MORE APPROPRIATE THAN CRACK GROWTH CRITERIA

<u>SUMMARY OF TEST RESULTS</u>

SPECIMEN NUMBER	ε_c (micro in/in)	ε_f	P_c (lbs)	P_f (lbs)	G_c (lb. in/in^2)	G_f
SL-1(C)	5160	5160	18520	18520	4.615	4.615
SL-2(T)	4249	5686	15200	20340	3.186	5.705
SL-3(T)	1768	5064	6320	18100	0.500	4.103
SL-4(T)	3528	4712	12640	16880	2.190	3.906
SL-5(C)	4810	4810	17220	17220	4.055	4.055
SL-6(C)	5228	5228	18600	18600	3.960	3.960
DL-1(T)	3282	4868	12000	17800	2.263	4.979
DL-2(T)	3086	4335	10960	15400	1.964	3.877
DL-3(T)	3101	4156	11180	14980	2.059	3.697
DL-4(C)	3965	3965	14500	14500	3.322	3.322

$$E_{effective} = 7.30 \times 10^6 \text{ psi}$$

("c" denotes starting critical value, "f" denotes final or failure)

PROCEDURES FOR DEMONSTRATING THE EFFECTS OF MATERIAL CHANGES ON THE STRUCTURAL RESPONSE OF COMPOSITE COMPONENTS

P. W. Mast

Naval Research Laboratory
Washington, DC 20375-5000

Material not received in time for publication.

THERMAL EXPANSION OF METAL MATRIX COMPOSITES

T. A. Hahn

Composite Materials Branch
Naval Research Laboratory
Washington, DC 20375-5000

ABSTRACT

Thermal deformation and stability of metal matrix composites have been the object of a number of recent studies [1,2] since these materials are being considered for potential applications where thermal loading will be important and since fabrication usually requires high processing temperatures. The mismatch in thermal expansion between the matrix and reinforcement leads to thermal stresses that not only effect the expansion but may also have considerable effect on the mechanical properties. A number of studies [3,4] have shown that the stresses generated in the ductile matrix material can reach the yield stress with only a small temperature change particularly at high temperatures where the yield stress is low. This results in permanent deformation of the composite and the dimensions to be a function of both temperature as well as past thermal history.

Tractable solid mixture models have been developed for both fiber [5] and eutectic composites [6] -the latter includes both the effect of work hardening and creep of the matrix although care must be exercised in applying this formulation [7]. Micromechanical models that take fiber geometry into consideration, have also been developed [3,4], however these formulations require finite element solutions.

The models presented in this study include both spherical and cylindrical geometry similar to that of Earmmee et al. [8] in the solution of plastic relaxation of misfit transformational strain of a spherical particle in an infinite matrix. The misfit strain is taken as the temperature change times the mismatch in thermal expansion coefficients of the matrix and reinforcement. The outside matrix diameter is variable according to the volume concentration of the composite. Step-wise temperature increments are used in the solution so that temperature dependent mechanical and thermal properties can be incorporated in the solutions. Detailed predicts of the expansion for both the spherical and cylindrical models will be compared to data on a modified Al-Si eutectic alloy, both unidirectional fiber and particulate SiC-Al composites and unidirectional W-Al composites.

REFERENCES

1. E, G. Wolff, "Thermal Expansion of Metal Matrix Composites," Report SD-TR-81-56, The Aerospace Corporation, El Segundo, California, August 1981.

2. B. A. Stein, S. S. Tompkins, and W. D. Brewer, "A Review of NASA Research on Low Temperature Metal Matrix Composites of Aeronautics and Space," Report MMC 200403, Proc. Fifth Metal Matrix Composites Technology Conf., NSWC, DoD Metal Matrix Composites Information Analysis Center, Santa Barbara, California, p. 14-1, May 1983.

3. G. J. Dvorak and M.S.M. Rao, "Thermal Stresses in Heat-Treated Fibrous Composites," Journal of Applied Mechanics, Vol. 98, p. 619, (1976).

4. M. H. Rice and G. A. Gurtman, "Residual Stresses in Fiber Reinforced Metal Matrix Composites," Final Report SSS-R-82-5447; Systems, Science, and Software; LoJolla, California, March 1982.

5. K. G. Kreider and V. M. Patarini, "Thermal Expansion of Boron Fiber-Aluminum Composites," Metallurgical Transactions, Vol. 1, p. 3431, (1970).

6. G. Garmong, "Elastic-Plactic Analysis of Deformation Induced by Thermal Stress in Eutectic Composites," Metallurgical Transactions, Vol. 5, pp. 2183-2191, (1974).

7. W. R. Tyson, "Discussion of Ref. [6]", Metallurgical Transactions, Vol. 6A, p. 1674, (1974).

8. Y. Y. Earmme, W. C. Johnson, and J. K. Lee, "Plastic Relaxation of the Transformation Strain Energy of a Misfitting Spherical Precipitate: Linear and Power-Law Strain Hardening," Metallurgical Transactions, Vol. 12A, p. 1521, (1981).

A MIXTURE MODEL FOR METAL-MATRIX COMPOSITES

H. Murakami and G. A. Hegemier

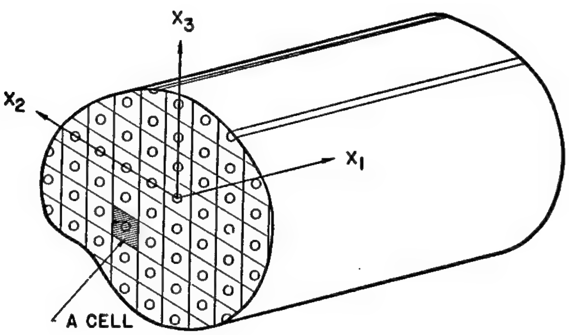
Department of Applied Mechanics and Engineering Sciences
University of California at San Diego

ABSTRACT

A methodology for constructing nonphenomenological nonlinear models for binary metal-matrix composites with initial periodic microstructure is discussed. The construction procedure is based upon the use of multivariable asymptotic representations and a modified Reissner-type variational principle. The resulting models assume the form of a binary mixture theory with microstructure. Current development has focused upon unidirectional fibrous composites. For such geometries, simulation capabilities include plastic deformation of the components and interface slip. From the global standpoint, these items translate into effective elastic moduli, effective plastic moduli, and degradation of these stiffnesses. In the case of dynamic problems, the models reflect geometric dispersion with respect to arbitrary propagation directions.

To-date a number of elastic validation problems have been examined. When compared with known exact data, the mixture simulations reveal accurate anisotropic global effective elastic moduli and geometric (first mode) dispersion. Based upon the results of such studies, the composite microstructural stress and deformation fields are thought to be sufficiently refined to provide an accurate simulation of the global anisotropic initial yield surface. Validation problems in the plastic regime are in progress to confirm this premise.

The present modeling technique is in response to the need for microstructural-based theories which furnish increased simulation capability in both the static and dynamic regimes with a minimum of model parameters to be experimentally determined.



FORMULATION

(a) Equations of motion

$$\sigma^{(\alpha)}_{ji,j} = \rho^{(\alpha)}\ddot{u}^{(\alpha)}_i \quad , \quad \sigma^{(\alpha)}_{ji} = \sigma^{(\alpha)}_{ij} \tag{1}$$

(b) Constitutive relations

$$\sigma^{(\alpha)}_{ij} = \lambda^{(\alpha)}\delta_{ij}e^{(\alpha)}_{kk} + 2\mu^{(\alpha)}e^{(\alpha)}_{ij} \qquad \text{(elastic)}$$

or

$$\dot{\sigma}_{ij} = D^{(\alpha)}_{ijkl}\,\dot{e}^{(\alpha)}_{kl} \qquad \text{(elasto-plastic)} \tag{2}$$

(c) Strain-displacement relations

$$e^{(\alpha)}_{ij} = \frac{1}{2}(\, u^{(\alpha)}_{i,j} + u^{(\alpha)}_{j,i}) \tag{3}$$

(d) Interface continuity conditions

$$\sigma^{(1)}_{ji}\nu^{(1)}_j = \sigma^{(2)}_{ji}\nu^{(1)}_j$$

$$u^{(1)}_i = u^{(2)}_i \quad \text{or} \quad u^{(2)}_i - u^{(1)}_i = [u_i] \tag{4}$$

(e) Initial conditions at t=0 and appropriate boundary conditions on ∂V.

MULTIVARIABLE REPRESENTATION

$$(x^*_2,\ x^*_3) = (x_2,\ x_3)/\varepsilon \tag{5}$$

$$f(x_i,t) = F(x_i,t;x^*_j;\varepsilon) \ ; \ i=1\text{-}3, j=II,III \tag{6}$$

x^*_J-periodicity condition

$$f_{,1} = F_{,1}\ ,\ f_{,2} = F_{,2} + F_{,II}/\varepsilon,\ f_{,3} = F_{,3} + F_{,III}/\varepsilon$$

or

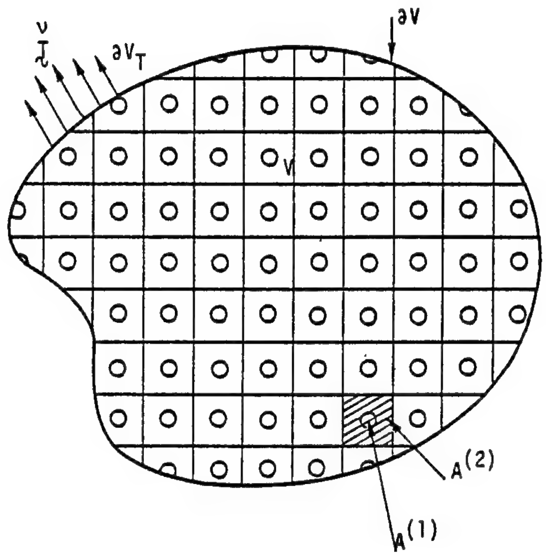
$$f_{,j} = F_{,j} + F_{,J}/\varepsilon \quad ; \ (\ )_{,I} = 0 \tag{7}$$

SYNTHESIZED FIELD RELATIONS

$$\sigma^{(\alpha)}_{ji,j} + \frac{1}{\varepsilon}\,\sigma^{(\alpha)}_{ji,J} = \rho^{(\alpha)}\ddot{u}^{(\alpha)}_i \tag{8}$$

$$\sigma^{(\alpha)}_{ij} = \lambda^{(\alpha)}\delta_{ij}e^{(\alpha)}_{kk} + 2\mu^{(\alpha)}e^{(\alpha)}_{ij} \tag{9}$$

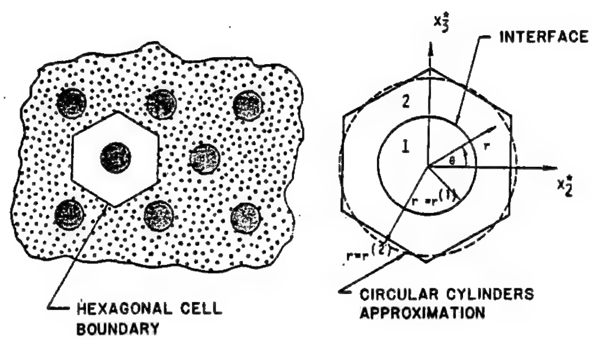
$$e^{(\alpha)}_{kl} = \frac{1}{2}\{u^{(\alpha)}_{k,l} + u^{(\alpha)}_{l,k} + \frac{1}{\varepsilon}(u^{(\alpha)}_{k,L} + u^{(\alpha)}_{l,K}\ )\} \tag{10}$$



VARIATIONAL PRINCIPLE FOR SYNTHESIZED FIELDS

Modified Reissner's stress-displacement variational principle (1984)

$$
\begin{aligned}
&\iiint_V \Big(\sum_{\alpha=1}^{2} \iint_{A^{(\alpha)}} \{ \delta e_{ij}^{(\alpha)} \sigma_{ij}^{(\alpha)} \\
&+ \sum_{kl \neq 11} \delta\sigma_{kl}^{(\alpha)} [\tfrac{1}{2}(u_{k,l}^{(\alpha)} + u_{l,k}^{(\alpha)} + \tfrac{1}{\varepsilon}(u_{k,L}^{(\alpha)} + u_{l,K}^{(\alpha)}) \} - e_{ij}^{(\alpha)} (\ldots)]\} dx_2^* dx_3^* \\
&+ \oint \{ \delta [u_i] \overset{\nu}{T}{}_i^* + \delta \overset{\nu}{T}{}_i^* [u_i] \} ds^* \Big) dx_1 dx_2 dx_3 \\
&= \iiint_V \Big(- \sum_{\alpha=1}^{2} \iint_{A^{(\alpha)}} \delta u_i^{(\alpha)} \rho^{(\alpha)} \ddot{u}_i^{(\alpha)} dx_2^* dx_3^* \Big) dx_1 dx_2 dx_3 \\
&+ \iint_{\partial V_T} \Big(\sum_{\alpha=1}^{2} \iint_{A^{(\alpha)}} \delta u_i^{(\alpha)} \overset{\nu}{T}{}_i^{(\alpha)} dx_2^* dx_3^* \Big) dS \qquad (11)
\end{aligned}
$$



DISPLACEMENT TRIAL FUNCTIONS

$$
u_1^{(\alpha)} = U_1^{(\alpha)}(x_k,t) + \varepsilon \overset{2}{S}_1(x_k,t) g_I^{(\alpha)}(r)\cos\theta + \varepsilon \overset{3}{S}_1(x_k,t) g_I^{(\alpha)}(r)\sin\theta
$$

$$
\begin{aligned}
u_2^{(\alpha)} = U_2^{(\alpha)} &+ \tfrac{\varepsilon}{2}(\overset{2}{S}_2 + \overset{3}{S}_3) g_I^{(\alpha)} \cos\theta \\
&+ \tfrac{\varepsilon}{2}(\overset{2}{S}_2 - \overset{3}{S}_3) \tilde{a}_2^{(\alpha)} [g_{II}^{(\alpha)} \cos\theta + g_{III}^{(\alpha)} \cos 3\theta] \\
&+ \tfrac{\varepsilon}{2} \overset{3}{S}_2 \tilde{a}_2^{(\alpha)} [g_{II}^{(\alpha)} \sin\theta + g_{III}^{(\alpha)} \sin 3\theta]
\end{aligned}
$$

$$
\begin{aligned}
u_3^{(\alpha)} = U_3^{(\alpha)} &+ \tfrac{\varepsilon}{2}(\overset{2}{S}_2 + \overset{3}{S}_3) g_I^{(\alpha)} \sin\theta \\
&- \tfrac{\varepsilon}{2}(\overset{2}{S}_2 - \overset{3}{S}_3) \tilde{a}_2^{(\alpha)} [g_{II}^{(\alpha)} \sin\theta - g_{III}^{(\alpha)} \sin 3\theta] \\
&+ \tfrac{\varepsilon}{2} \overset{3}{S}_2 \tilde{a}_2^{(\alpha)} [g_{II}^{(\alpha)} \cos\theta - g_{III}^{(\alpha)} \cos 3\theta] \qquad (12)
\end{aligned}
$$

where

$$
\overset{j}{S}_i = \frac{1}{\varepsilon A} \oint u_i^{(\alpha)} \nu_j^{(\alpha)} ds^* \quad ; \quad \overset{3}{S}_2 = \overset{2}{S}_3 \qquad (13)
$$

STRESS TRIAL FUNCTIONS

$$\sigma_{12}^{(\alpha)} = \tau_{12}^{(\alpha)}(x_k,t) + \delta_{\alpha 2} r^{-2}\{t_{12}^{(\alpha)}(x_k,t)\cos 2\theta - t_{31}^{(\alpha)}(x_k,t)\sin 2\theta\}$$
$$+ \frac{\varepsilon}{2} P_1(x_k,t) g_I^{(\alpha)}(r)\cos\theta$$

$$\sigma_{31}^{(\alpha)} = \tau_{31}^{(\alpha)}(x_k,t) + \delta_{\alpha 2} r^{-2}\{t_{12}^{(\alpha)}(x_k,t)\sin 2\theta + t_{31}^{(\alpha)}(x_k,t)\cos 2\theta\}$$
$$+ \frac{\varepsilon}{2} P_1(x_k,t) g_I^{(\alpha)}(r)\sin\theta$$

$$\sigma_{22}^{(\alpha)} = \tau_{22}^{(\alpha)} + \delta_{\alpha 2} r^{-2} t_{22}^{(\alpha)}\cos 2\theta + t_{33}^{(\alpha)}(f_I^{(\alpha)} - f_{II}^{(\alpha)}\cos 2\theta + f_{III}^{(\alpha)}\cos 4\theta)$$
$$-t_{23}^{(\alpha)}(f_{II}^{(\alpha)}\sin 2\theta - f_{III}^{(\alpha)}\sin 4\theta) - \varepsilon P_2(h_I^{(\alpha)}\cos\theta - h_{III}^{(\alpha)}\cos 3\theta)$$
$$-\varepsilon P_3(h_{II}^{(\alpha)}\sin\theta - h_{III}^{(\alpha)}\sin 3\theta)$$

STRESS TRIAL FUNCTIONS

$$\sigma_{33}^{(\alpha)} = \tau_{33}^{(\alpha)} - \delta_{\alpha 2} r^{-2} t_{22}^{(\alpha)}\cos 2\theta - t_{33}^{(\alpha)}(f_I^{(\alpha)} + f_{II}^{(\alpha)}\cos 2\theta + f_{III}^{(\alpha)}\cos 4\theta)$$
$$-t_{23}^{(\alpha)}(f_{II}^{(\alpha)}\sin 2\theta + f_{III}^{(\alpha)}\sin 4\theta) - \varepsilon P_2(h_{II}^{(\alpha)}\cos\theta + h_{III}^{(\alpha)}\cos 3\theta)$$
$$-\varepsilon P_3(h_I^{(\alpha)}\sin\theta + h_{III}^{(\alpha)}\sin 3\theta)$$

$$\sigma_{23}^{(\alpha)} = \tau_{23}^{(\alpha)} + \delta_{\alpha 2} r^{-2} t_{22}^{(\alpha)}\sin 2\theta + t_{33}^{(\alpha)} f_{III}^{(\alpha)}\sin 4\theta$$
$$+t_{23}^{(\alpha)}(f_I^{(\alpha)} - f_{III}^{(\alpha)}\cos 4\theta) - \varepsilon P_2(h_{IV}^{(\alpha)}\sin\theta - h_{III}^{(\alpha)}\sin 3\theta)$$
$$-\varepsilon P_3(h_{IV}^{(\alpha)}\cos\theta + h_{III}^{(\alpha)}\cos 3\theta) \tag{14}$$

where

$$P_i = \frac{1}{\varepsilon A}\int \sigma_{ji}^{(\alpha)} \nu_j^{(1)} ds^* \tag{15}$$

ASYMPTOTIC ANALYSIS

$$\{u_i,\ \sigma_{ij}\}^{(\alpha)}(x_k, x_J^*, t; \varepsilon) = \sum_{n=0}^{N} \varepsilon^n \{u_{i(n)}, \sigma_{ij(n)}\}^{(\alpha)}(x_k, x_J^*, t) \tag{16}$$

$$O(\varepsilon^{-1}): \quad \sigma_{2i(0),II}^{(\alpha)} + \sigma_{3i(0),III}^{(\alpha)} = 0 \tag{17}$$

$$u_{k(0),L}^{(\alpha)} + u_{l(0),K}^{(\alpha)} = 0 \tag{18}$$

$O(\varepsilon^n)$, $n \geq 0$:

$$\sigma_{2i(n+1),II}^{(\alpha)} + \sigma_{3i(n+1),III}^{(\alpha)} = \rho^{(\alpha)} \ddot{u}_{i(n)}^{(\alpha)} - \sigma_{1i(n),1}^{(\alpha)} \tag{19}$$

$$\sigma_{ij(n)}^{(\alpha)} = \lambda^{(\alpha)} \delta_{ij} e_{kk(n)}^{(\ddot{a})} + 2\mu^{(\alpha)} e_{ij(n)}^{(\alpha)} \tag{20}$$

$$e_{kl(n)}^{(\alpha)} = \frac{1}{2}(u_{k(n),l}^{(\alpha)} + u_{l(n),k}^{(\alpha)} + u_{k(n+1),L}^{(\alpha)} + u_{l(n+1),K}^{(\alpha)}) \tag{21}$$

$$u_{i(0)}^{(1)} = u_{i(0)}^{(2)} = u_{i(0)}(x_k,t) \tag{22}$$

$$\sigma_{2i(1),II}^{(\alpha)} + \sigma_{3i(1),III}^{(\alpha)} = (-1)^{\alpha+1} P_i / n^{(\alpha)} \tag{23}$$

MIXTURE EQUATIONS OF MOTION

$$(n^{(\alpha)}\sigma_{ji}^{(\alpha a)})_{,j} + (-1)^{\alpha+1} P_i = n^{(\alpha)} \rho^{(\alpha)} \ddot{U}_i^{(\alpha)} \tag{24}$$

$$\overset{2}{M}_{11,1} + \overset{2}{M}_{21,2} + \varepsilon^{-2}(\sigma_{12}^{(2a)} - \sigma_{12}^{(1a)} - b_I \overset{2}{S}_1) = I_{11} \ddot{\overset{2}{S}}_1$$

$$\overset{3}{M}_{11,1} + \overset{3}{M}_{31,3} + \varepsilon^{-2}(\sigma_{31}^{(2a)} - \sigma_{31}^{(1a)} - b_I \overset{3}{S}_1) = I_{11} \ddot{\overset{3}{S}}_1$$

$$\overset{2}{M}_{12,1} + \overset{2}{M}_{22,2} + \overset{2}{M}_{32,3}$$
$$+ \varepsilon^{-2}[\sum_{\alpha=1}^{2}(-1)^{\alpha}\{(\sigma_{22}^{(\alpha a)} + \sigma_{33}^{(\alpha a)})/2 + b_{II}^{(\alpha)}(\sigma_{22}^{(\alpha a)} - \sigma_{33}^{(\alpha a)})\}$$
$$-b_I(\overset{2}{S}_2 + \overset{3}{S}_3) + b_{III}(\overset{2}{S}_2 - \overset{3}{S}_3)] = I_{22}\ddot{\overset{2}{S}}_2 + I_{23}\ddot{\overset{3}{S}}_3$$

$$\overset{3}{M}_{13,1} + \overset{3}{M}_{23,2} + \overset{3}{M}_{33,3}$$
$$+ \varepsilon^{-2}[\sum_{\alpha=1}^{2}(-1)^{\alpha}\{(\sigma_{22}^{(\alpha a)} + \sigma_{33}^{(\alpha a)})/2 - b_{II}^{(\alpha)}(\sigma_{22}^{(\alpha a)} - \sigma_{33}^{(\alpha a)})\}$$
$$-b_I(\overset{2}{S}_2 + \overset{3}{S}_3) - b_{III}(\overset{2}{S}_2 - \overset{3}{S}_3)] = I_{32}\ddot{\overset{2}{S}}_2 + I_{33}\ddot{\overset{3}{S}}_3$$

$$\overset{3}{M}_{22,2} + \overset{2}{M}_{33,3} + \varepsilon^{-2}\{\sum_{\alpha=1}^{2}(-1)^{\alpha}\sigma_{23}^{(\alpha a)} b_{IV}^{(\alpha)} + b_V \overset{3}{S}_2\} = I_{44}\ddot{\overset{3}{S}}_2 \tag{25}$$

NONLINEAR UNIDIRECTIONAL METAL MATRIX COMPOSITE BEHAVIOR

K. W. Buesking

Materials Sciences Corporation
Gwynedd Plaza II
Bethlehem Pike
Spring House, PA 19477

ABSTRACT

Metal matrix composites, especially graphite/aluminum, appear to be very attractive for satellite structures because of their high ratio of stiffness to density and their near zero thermal expansion coefficient. However, experimental measurements of axial strains in unidirectional graphite/aluminum induced by orbital thermal cycles show that the thermal expansion of the composite is highly nonlinear. The nonlinearity has been attributed to micromechanical stresses within the composite which are sufficiently large to cause yielding within the matrix. Matrix plasticity will strongly affect not only the composite thermal expansion but also other composite properties especially the transverse extensional modulus and the axial shear modulus. Furthermore, the mechanical response of a composite with a plastic matrix will be history dependent. Two identical pieces of material at the same stress state and temperature may exhibit completely different properties which are a result of their previous loading history.

This paper first outlines previous studies by other investigators who have examined the nonlinear behavior of metal matrix composites. The previous studies can be grouped into either detailed finite element analyses or models based upon the approximate stress fields related to the series and parallel rule of mixtures. The paper then describes an alternate approximate material model which is termed the phase average stress model [1]. This model uses properties of the fiber, matrix and unidirectional composite (determined from the composite cylinders assemblage) to compute average stresses within the fiber and matrix. The matrix stresses are then used in a von Mises yield function to determine the composite load at which the matrix yields. A flow rule for the matrix, based upon assumptions of a kinematic linear work hardening material [2], is used to define an incremental effective matrix compliance during yielding. The effective matrix compliance is then used in the composite cylinders assemblage to define the incremental composite response. Therefore, any combination of composite stress and temperature history can be applied incrementally to the phase average stress model to determine the nonlinear response of the unidirectional material.

The paper presents comparisons of results computed using the phase average stress model with previous finite element models, other approximate models and with measured material behavior. The comparisons appear to be reasonable within the limitations of the modelling assumptions.

The phase average stress model is also exercised to examine potential solutions to the thermal hysteresis problem. The theoretical results suggest that the hysteresis may be reduced by using a magnesium matrix, or possibly eliminated by increasing the in situ yield strength of the aluminum matrix.

It is concluded that the concept of phase average stresses is a fairly simple yet accurate method for studying nonlinear behavior of unidirectional composite materials. It would be relatively easy to incorporate the unidirectional material model in a nonlinear laminate or structural analysis code. The concept of phase average stresses is not limited to unidirectional fibers or matrix plasticity. It may be applied to any type of two phase composite whose effective properties are known and can also be examined for other matrix effects such as creep.

REFERENCES

1. Hashin, Z. and Humphreys, E. A., "Elevated Temperature Behavior of Metal Matrix Composites," MSC TFR on AFOSR Contract F49620-79-C-0059, November, 1981.

2. Martin, J. B., Plasticity: Fundamentals and General Results, MIT Press, Cambridge, MA, 1975.

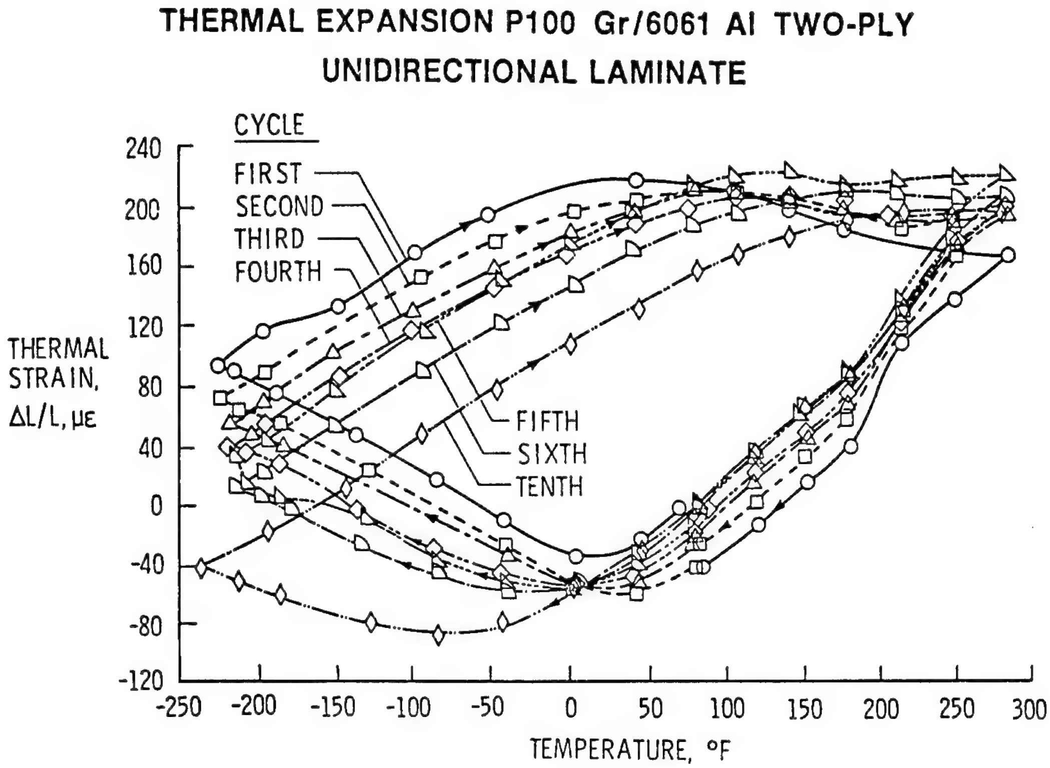
NONLINEAR UNIDIRECTIONAL METAL MATRIX COMPOSITE BEHAVIOR

K. W. BUESKING

WORK PERFORMED UNDER NSWC CONTRACT NO.
N60921-83-C-0155

OBJECTIVE

- MODEL ELASTIC PLASTIC BEHAVIOR OF UNIDIRECTIONAL METAL MATRIX COMPOSITES
- EXPLAIN MEASURED MATERIAL BEHAVIOR
- ACCOUNT FOR GENERAL LOADING, HISTORY DEPENDENCE, ETC.
- SUBSEQUENTLY USE MODEL IN NONLINEAR LAMINATE ANALYSIS, STRUCTURAL ANALYSIS AND MATERIAL IMPROVEMENT STUDIES



Thermal Cycling Hysteresis seen in P100/6061 Unidirectional Composites. Data obtained from Steve Tompkins - NASA Langley.

BACKGROUND

- PREVIOUS FINITE ELEMENT STUDIES
  - ADAMS, JCM 1970
  - DVORAK, ET AL., JCM 1973
  - FOYE, JCM 1973
  - HASHIN AND HUMPHREYS, AFOSR, 1981
- PREVIOUS APPROXIMATE STRESS FIELD STUDIES
  - HUANG, JCM 1971
  - BAHEI EL-DIN AND DVORAK, ASME 1979, JAM 1982, JAM 1982
  - DVORAK, IUTAM 1983
  - MIN, J MECH. AND PHYS. SOLIDS 1981
  - MIN AND CROSSMAN, ASTM STP 1982, JCM 1982

PHASE AVERAGE STRESS MODEL

- ASSUME
  - ACTUAL VARIABLE STRESS FIELD IN FIBERS AND MATRIX CAN BE APPROXIMATED BY UNIFORM AVERAGE STRESS

- APPROACH
  - GIVEN FIBER AND MATRIX PROPERTIES
  - COMPUTE UNIDIRECTIONAL COMPOSITE PROPERTIES FROM COMPOSITE CYLINDER ASSEMBLAGE MODEL (HASHIN, JAM 1979)
  - COMPUTE AVERAGE STRESSES IN FIBER AND MATRIX

PHASE AVERAGE STRESS MODEL

- DEFINE AVERAGE STRESSES AND STRAINS

$$v_f \, d\sigma_{ij}^{f} + v_m \, d\sigma_{ij}^{m} = d\sigma_{ij}^{*}$$

$$v_f \, d\varepsilon_{ij}^{f} + v_m \, d\varepsilon_{ij}^{m} = d\varepsilon_{ij}^{*}$$

- STRESS-STRAIN RELATIONS

$$d\varepsilon_{ij}^{f} = S_{ijk\ell}^{f} \, d\sigma_{ij}^{f} + \alpha_{ij}^{f} \, dT$$

$$d\varepsilon_{ij}^{m} = S_{ijk\ell}^{m} \, d\sigma_{ij}^{m} + \alpha_{ij}^{m} \, dT$$

$$d\varepsilon_{ij}^{*} = S_{ijk\ell}^{*} \, d\sigma_{ij}^{*} + \alpha_{ij}^{*} \, dT$$

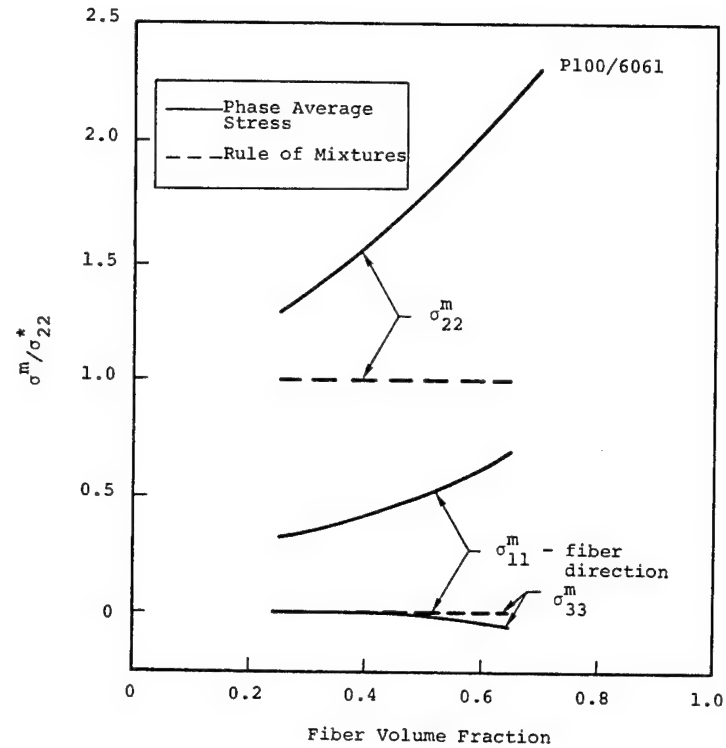
- SUBSTITUTING

$$d\sigma_{ij}^{m} = \frac{1}{V_m} \; S_{ijk\ell}^{I} \left[(S_{k\ell mn}^{f} - S_{k\ell mn}^{*}) d\sigma_{mn}^{*} + (v_f \alpha_{k\ell}^{f} + v_m \alpha_{k\ell}^{m} - \alpha_{k\ell}^{*}) \; dT \right]$$

WHERE,

$$S_{ijk\ell}^{I} \; (S_{k\ell mn}^{f} - S_{k\ell mn}^{m}) = I_{ijmn} = \frac{1}{2} \; (\delta_{im} \, \delta_{jn} + \delta_{in} \, \delta_{jm})$$

AVERAGE MATRIX STRESSES DUE TO TRANSVERSE LOAD



COMPOSITE YIELD SURFACE

- ASSUME MATRIX OBEYS VON MISES YIELD CONDITION:

$$\phi = (s^m_{ij} - he^{mp}_{ij})\,(s^m_{ij} - he^{mp}_{ij}) - 3k^2$$

WHERE

$$s^m_{ij} = \sigma^m_{ij} - \frac{1}{3}\,\sigma^m_{kk}\,\delta_{ij} \quad \text{(DEVIATORIC STRESS)}$$

$$e^{mp}_{ij} = \varepsilon^{mp}_{ij} - \frac{1}{3}\,\varepsilon^{mp}_{kk}\,\delta_{ij} \quad \text{(PLASTIC STRAIN, = 0 FOR INITIAL YIELD)}$$

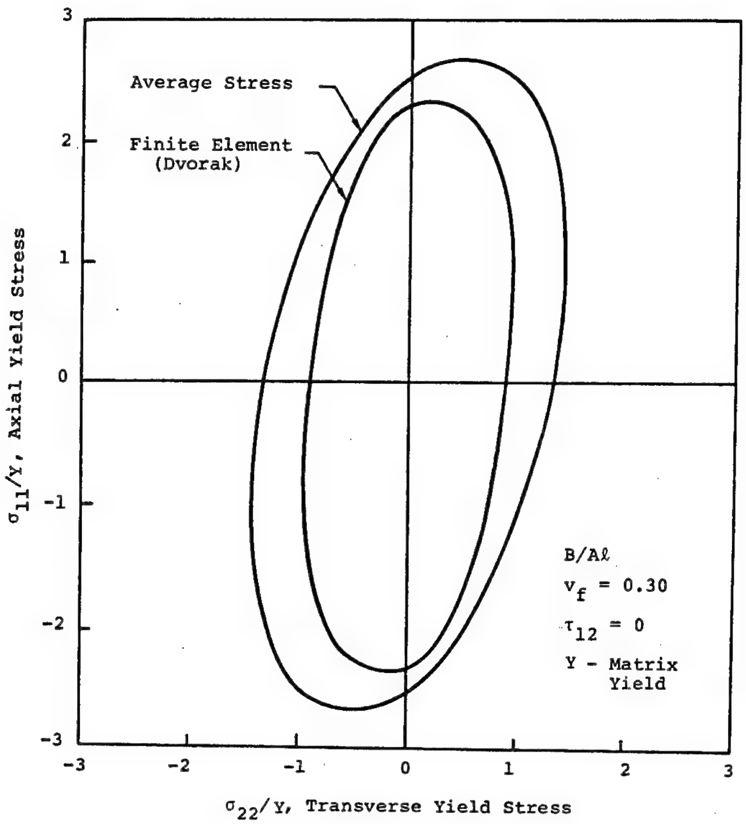
$$h = \frac{2}{3}\,\frac{E^m E^m_t}{E^m - E^m_t} \quad \text{(PLASTIC MODULUS)}$$

$$k^2 = \frac{2}{9}\,(\sigma^m_y)^2 \quad \text{(YIELD STRESS)}$$

- COMPOSITE YIELDS WHEN AVERAGE MATRIX STRESSES SATISFY YIELD CONDITION

$$\phi = 0$$

COMPARISON OF COMPOSITE YIELD SURFACES



POST YIELD RESPONSE

- ASSUME MATRIX IS A KINEMATIC LINEAR WORK HARDENING MATERIAL

- YIELD SURFACE

$$\phi = (s^m_{ij} - he^{mp}_{ij})\ (s^m_{ij} - he^{mp}_{ij}) - 3k^2$$

- PLASTIC STRAIN INCREMENT

$$de^{mp}_{ij} = \frac{1}{3hk^2}\ (s^m_{ij} - he^{mp}_{ij})\ (s^m_{k\ell} - he^{mp}_{k\ell})\ ds^m_{k\ell}$$

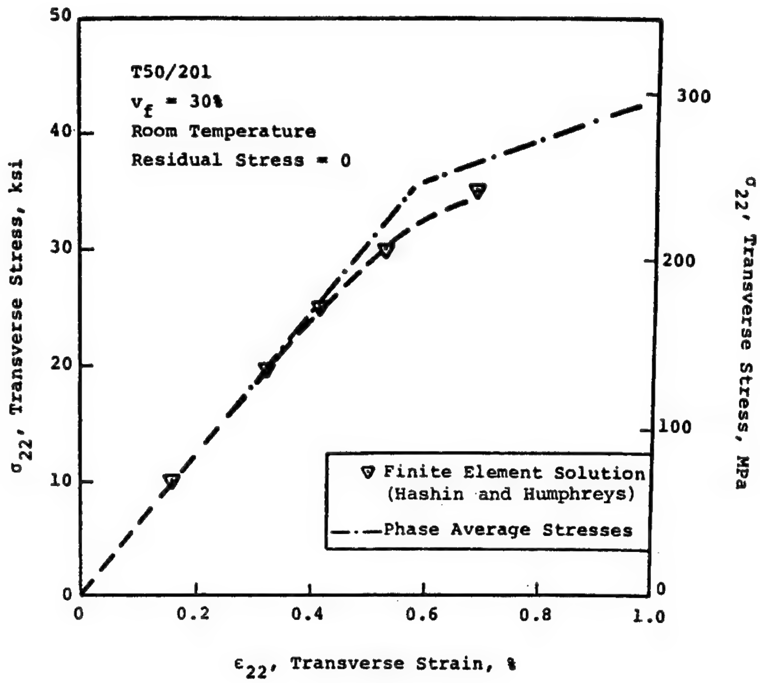
- TOTAL MATRIX STRAIN

$$d\varepsilon^m_{ij} = d\varepsilon^{me}_{ij} + d\varepsilon^{mp}_{ij} + d\varepsilon^{mt}_{ij}$$

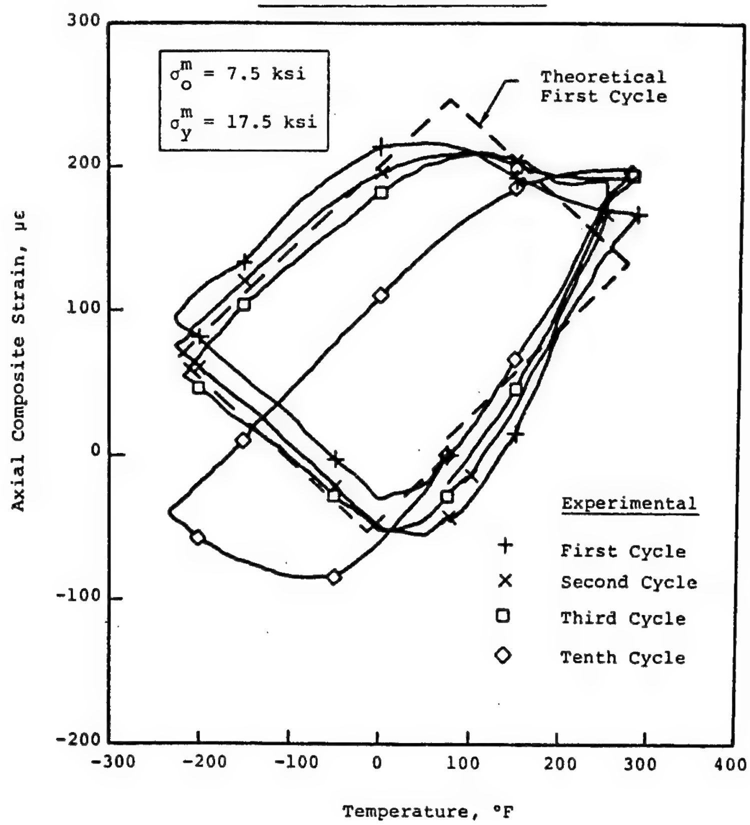
$$d\varepsilon^m_{ij} = (S^{me}_{ijk\ell} + S^{mp}_{ijk\ell})\ d\sigma^m_{k\ell} + \alpha^m_{ij}\ dT$$

- FOR COMPOSITE INCREMENT, ASSUME MATRIX IS PIECEWISE LINEAR MATERIAL WITH PROPERTIES DEFINED BY SUM OF ELASTIC AND PLASTIC COMPLIANCES

TRANSVERSE STRESS-STRAIN



P100/6061 THERMAL STRAINS



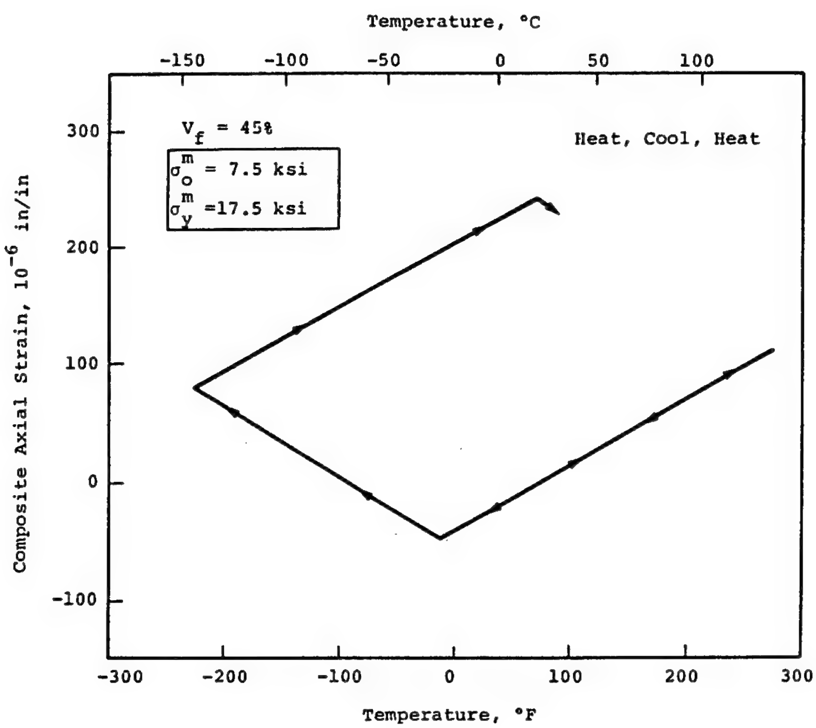
P100/6061 PROPERTIES DURING THERMAL CYCLE

Computed Axial Strains, Composite Properties and Matrix Stresses in P100/Al Unidirectional Material During Initial Thermal Hysteresis Cycle

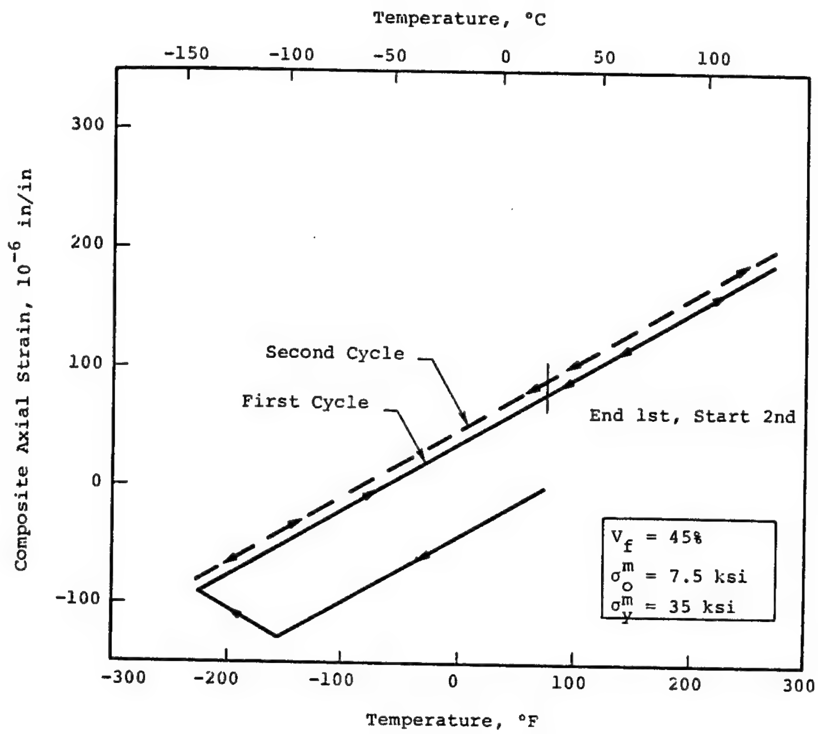
Temp °F	ϵ_a^* 10^{-6}in/in	E_a^* Msi	E_t^* Msi	G_a^* Msi	ν_a^* --	ν_t^* --	α_a^* 10^{-6}/°F	α_t^* 10^{-6}/°F	σ_a^m ksi	σ_t^m ksi	Matrix State
75	0.0	50.6	3.80	3.15	0.338	0.392	0.556	15.7	7.50	0.0	Elastic
-9.4	-46.9	50.6	3.80	3.15	0.338	0.392	0.556	15.7	17.8	0.28	Yield
-125	20.3	46.1	1.23	1.15	0.451	0.592	-0.568	17.2	21.0	0.56	Plastic
-225	78.4	46.1	1.23	1.15	0.451	0.592	-0.568	17.2	23.8	0.80	Plastic
-125	134	50.6	3.80	3.15	0.338	0.392	0.556	15.7	11.7	0.467	Elastic
70.4	243	50.6	3.80	3.15	0.338	0.392	0.556	15.7	-12.2	-0.186	Yield
175	182	46.1	1.23	1.15	0.451	0.592	-0.568	17.2	-15.1	-0.438	Plastic
275	124	46.1	1.23	1.15	0.451	0.592	-0.568	17.2	-17.9	-0.679	Plastic
175	68.1	50.6	3.80	3.15	0.338	0.392	0.556	15.7	-5.71	-0.344	Elastic
75	12.5	50.6	3.80	3.15	0.338	0.392	0.556	15.7	6.47	-10.3	Elastic

Note: 45% Fibers, Residual Stress = 7.5 ksi, Matrix Yield Strength 17.5 ksi, ρ = 0.087 lb/in^3

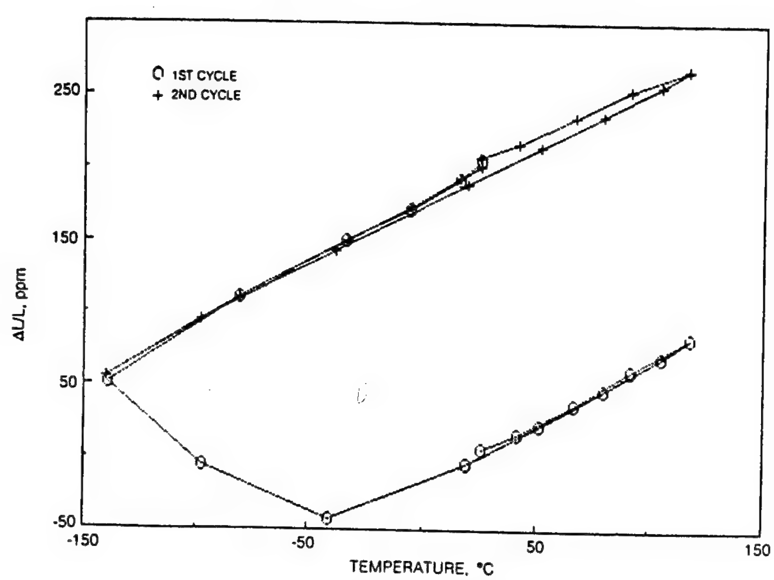
REVERSED THERMAL CYCLE



HIGH STRENGTH MATRIX

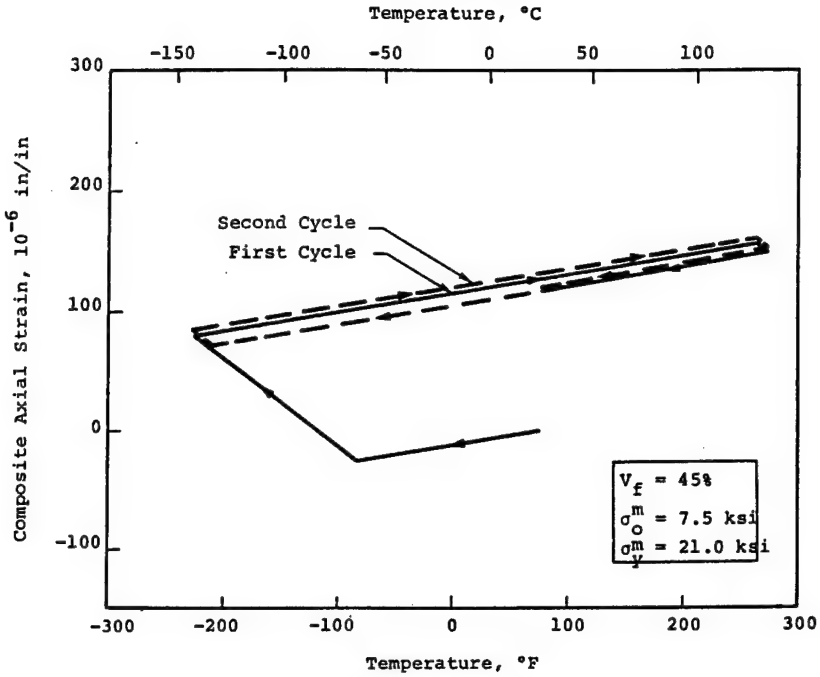


±10° LAMINATE - HIGH STRENGTH MATRIX



Experimental Data from Mayor and Strife, UTRC

P100/MG THERMAL STRAINS



CONCLUSIONS

- PHASE AVERAGE STRESS MODEL PROVIDES SIMPLE, YET FAIRLY ACCURATE TOOL FOR STUDYING NONLINEAR METAL MATRIX COMPOSITE BEHAVIOR.

- THERMAL HYSTERESIS MAY BE IMPROVED BY:
  - HIGHER STRENGTH ALUMINUM MATRIX
  - USE OF MAGNESIUM MATRIX

APPENDIX A: PROGRAM LISTINGS

AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
MATERIALS LABORATORY

INHOUSE

ADVANCED COMPOSITES
WORK UNIT DIRECTIVE (WUD) NUMBER 45
77 April - 85 April

WUD Leader: James M. Whitney
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autovon: 785-6685

Objective: The objective of the current thrust under this work is to develop and demonstrate concepts of damage resistance as applied to fiber reinforced composite laminates. Short term objectives (1-2 yrs) include the following:

(a) Development of failure mode models with emphasis on delamination and matrix cracking.
(b) Assess the role of matrix toughness in composite failure processes.
(c) To develop concepts of interface/interphase strengthening.

CONTRACTS

IMPROVED, DAMAGE RESISTANT COMPOSITE MATERIALS
F33615-84-C-5070
1 Sep 84 - 1 Feb 88

Project Engineer: James M. Whitney
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autovon: 785-6685

Principal Investigator: Ron Servais
University of Dayton Research Institute
300 College Park Avenue
Dayton, OH 45469

Objective: The objective of this program is to investigate from both an experimental and analytical standpoint the potential of new and/or modifications of existing polymeric materials and reinforcement forms for use in advanced composite materials, including processing/mechanical property relationships. Such materials are subsequent candidates for use in advanced aircraft and aerospace structural applications.

FAILURE RESISTANT COMPOSITE CONCEPTS—IMPROVED POST-BUCKLING BEHAVIOR
F33615-83-K-5016
1 Jun 83 - 30 Nov 85

Project Engineer: James M. Whitney
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autovon: 785-6685

Principal Investigator: James W. Mar
Technology Laboratory for Advanced Composites
Dept of Aeronautics and Astronautics
Massachusetts Institute of Technology
Cambridge, MA 02139

Objective: The objective of this program is to develop materials concepts for improving the postbuckling behavior of laminated plates and cylindrical shells for application to aircraft structures. Program involves both analytical and experimental work.

CUMULATIVE DAMAGE MODEL FOR COMPOSITE MATERIALS
F33615-80-C-5039
23 Feb 81 - Apr 85

Project Engineer: Marvin Knight
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-7131 Autovon: 785-7131

Principal Investigator: P. C. Chou
Dyna East Corporation
227 Hemlock Road
Wynnewood, P 19096
(215) 895-2288

Objective: This program will develop a methodology, including analytical modeling, for predicting and experimentally characterizing advanced composite materials' mechanical responses to defined load histories. A cumulative damage model is the ultimate goal.

CUMULATIVE DAMAGE MODEL FOR COMPOSITE MATERIALS
F33615-81-C-5049
23 Feb 81 - Apr 85

Project Engineer: Marvin Knight
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-7131 Autovon: 785-7131

Principal Investigator: H. Miller
General Dynamics Corporation
Fort Worth Division
P.O. Box 748
Fort Worth, TX 76101
(817) 732-4811 Ext 5375

Objective: This program will develop a methodology, including analytical modeling, for predicting and experimentally characterizing advanced composite materials' mechanical responses to defined load histories. A cumulative damage model is the ultimate goal.

FUNDAMENTAL MATRIX STIFFNESS FORMULATIONS FOR LAMINATE STRUCTURES
F33615-83-C-5076
1 Jun 83 - 31 Mar 86

Project Engineer: Steven L. Donaldson
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-6685 Autovon: 785-6685

Principal Investigator: Henry T. Yang
School of Aeronautical & Astronautical Engineering
Purdue University
West Lafayette, IN 47907
(317) 494-5117

Objective: This program will develop the mathematical formulation of the stiffness matrices of laminated plates and beams, to ultimately obtain the stress fields, the vibrational, and the buckling response of structural laminates. The elements will include the provision to handle individual failed plies or delaminations. The elements will be formulated in such a way that they can be simply implemented on micro and minicomputers.

FAILURE ANALYSIS FOR COMPOSITE STRUCTURE MATERIALS
F33615-84-C-5010
Jun 84 - Nov 86

Project Engineer: Frank Fechek
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLSE
Wright-Patterson AFB, OH 45433
(513) 255-7481 Autovon: 785-7481

Principal Investigator: Brian Smith
The Boeing Company
P.O. Box 3707
Mail Stop 73-43
Seattle, WA 98124

Objective: The objectives of this program are: a) to verify the capability of state-of-the-art analysis techniques and procedures to produce useful data toward the understanding of the cause of composite failures, beginning with the failed part and, b) to organize this information into a compendium defining a failure logic network which will assist the failure analyst in sequentially selecting the appropriate tests, techniques, and procedures to be applied when conducting a failure analysis of a composite structure.

CURING PROCESS OF COMPOSITE MATERIALS
F33615-84-C-5049
Sep 84 - 1 Oct 87

Project Engineer: Stephen W. Tsai
Materials Laboratory
Air Force Wright Aeronautical Laboratories
AFWAL/MLBM
Wright-Patterson AFB, OH 45433
(513) 255-3068 Autovon: 785-3068

Principal Investigator: George S. Springer
Dept of Aeronautics and Astronautics
Stanford University
Stanford, CA 94305
(415) 497-4135

Objective: To extend the analytical modeling developed by the Principal Investigator to include the curing thermosetting and thermal plastics as the matrix materials. To provide criteria for automated process controls and optimization.

FLIGHT DYNAMICS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES

IN-HOUSE

STRUCTURAL INTEGRITY RESEARCH FOR METALS AND COMPOSITES
JON: 2307N124
83 September 30 - 85 October 15

Project Engineer: Dr. G. P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Objective: To resolve theoretical questions and develop damage tolerance and life analysis methods which can be used to satisfy the requirements of MIL-STD-1530A for advanced composite and metallic airframe structures. The specific objectives in the composites area are:

(a) perform a critical assessment of the literature on damage tolerance and effects-of-defects in resin-matrix composite structures;

(b) verify the nonlinear single mode analytical results by using structural response data for simple rectangular graphite-epoxy panels loaded by low and high level broadband random acoustic excitation; and

(c) develop a basic understanding of the damage accumulation mechanisms in and the factors that affect the acoustic fatigue life of the composite cantilever plate specimens used in-house.

ACOUSTIC FATIGUE DESIGN OF ADVANCED STRUCTURES
JON: 24010146
82 February 3 - 85 February 2

Project Engineer: Howard F. Wolfe
Air Force Wright Aeronautical Laboratories
AFWAL/FIBED
Wright-Patterson AFB, Ohio 45433
(513) 255-5753 Autovon 785-5753

Objective: Develop methods for the prediction of the acoustical fatigue life of graphite-epoxy skin stringer structures. The investigation will include adhesive bonded and co-cured skin stringer beams and acoustic panels to be tested on a vibration shaker and in a progressive wave tube to obtain dynamic properties and fatigue life.

REPAIR OF GRAPHITE/EPOXY COMPOSITES
JON: 24010344
84 March 1 - 85 February 1

Project Engineer: Forrest Sandow
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCB
Wright-Patterson AFB, Ohio 45433
(513) 255-2582 Autovon 785-2582

Objective: At present the development of repair procedures for graphite epoxy composites is generally based on the testing of simulated repairs which are large and relatively costly to produce and evaluate. This problem should be overcome by the use of elemental joint specimens representing a section through the repaired region. Detailed studies of the stress-strain behavior of these specimens, under the appropriate variables, will provide basic information for design of repairs.

ASSESSMENT OF CORROSION CONTROL PROTECTION COATINGS
JON: 24010350
80 April 28 - 85 May 1

Project Engineer: Billy White
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCA
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-5864

Objective: To determine the susceptibility of graphite/epoxy-aluminum joints to corrosion when protective coatings, that have undergone fatigue loading, are used. The knowledge gained will be used to determine if present corrosion control systems actually prevent corrosion and if not, how they could be modified to prevent corrosion from occurrung.

CONTRACTS

A STUDY OF THE BUCKLING, POST-BUCKLING BEHAVIOR AND VIBRATION OF LAMINATED COMPOSITE PLATES
Contract F33615-81-K-3203 JON: 2307N115
80 November 20 - 85 January 20

Project Engineer: Dr. N. S. Khot
Air Force Wright Aeronautical Laboratories
AFWAL/FIRBA
Wright-Patterson AFB, Ohio 45433
(513) 255-6992 Autovon 785-6992

Principal Investigator: Professor Arthur Leissa
Department of Engineering Mechanics
Ohio State University
155 West Woodruff Avenue
Columbus, Ohio 43210

Objective: To prepare a monograph summarizing the state of the art in buckling, post-buckling and vibration behavior of laminated composite plates.

FATIGUE CRACK RETARDATION IN A METAL-MATRIX COMPOSITE DUE TO OVERLOADS
Contract F33615-82-K-3218 JON: 2307N120
82 June 15 - 85 May 31

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Dr. C. T. Sun
School of Aeronautics and Astronautics
Purdue University
West Lafayette, Indiana 47907
(317) 494-5130

Objective: The objective is to develop an understanding of and analytical procedures for predicting the effects of overloads on crack growth in fiber-reinforced metal-matrix composite materials.

EFFECTS OF POROSITY ON DELAMINATION OF RESIN-MATRIX COMPOSITES
Contract F33615-84-C-3205 JON: 2307N125
84 June 04 - 87 June 30

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

DURABILITY OF CONTINUOUS FIBER REINFORCED METAL METRIX COMPOSITES
Contract F33615-83-C-3219 JON: 24010167
83 September 1 - 87 February 6

Project Engineer: Frank M. Grimsley
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: C. R. Saff
McDonnell Aircraft Co.
St. Louis, Missouri 63166
(314) 234-1594

Objective: Develop structural life analysis methods for continuous fiber-reinforced metal-matrix composite materils subjected to constant amplitude fatigue loading.

EFFECT OF FREQUENCY AND STACKING SEQUENCE ON THE TENSILE FATIGUE BEHAVIOR OF COMPOSITE MATERIALS
Contract F33615-83-K-3220 JON: 24010171
83 May 18 - 85 May 12

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Dr. V. Sarma Avva
Mechanical Engineering Department
North Carolina A&T State University
Greensboro, North Carolina 27411
(919) 379-7620

Objective: Assess the effect of test frequency and stacking sequence on fatigue life of resin-matrix composite materials.

ADHESIVE STRIP CONCEPT FOR DELAMINATION ARRESTMENT
Contract F33615-84-C-3201 JON: 24010183
84 August 1 - 85 November 1

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Dr. J. Eisenmann
General Dynamics Corporation
P.O. Box 748
Fort Worth, Texas 76101

Objective: To evaluate an adhesive strip concept for preventing the occurrance of delaminations in resin-matrix composite structures and for arresting their growth if they occur.

DESIGN VERIFICATION FOR OPTIMIZED PANELS
Contract F33615-81-C-3222 JON: 24010248
81 September 15 - 85 September 30

Project Engineer: Dr. N. S. Khot
Air Force Wright Aeronautical Laboratories
AFWAL/FIBRA
Wright-Patterson AFB, Ohio 45433
(513) 255-6992 Autovon 785-6992

Principal Investigator: Professor John F. Mandell
Department of Materials Science and Engineering
Massachusetts Institute of Technology
Cambridge, Massachusetts 02139
(617) 253-7181

Objective: To develop a basic understanding, models and analyses of the effect of porosity on delamination growth in resin-matrix composite materials.

DETECTION OF FAILURE PROGRESSION IN CROSSPLY GRAPHITE/EPOXY DURING FATIGUE LOADING THROUGH ACOUSTIC EMISSION
Contract F33615-84-C-3204 JON: 2307N126
84 June 1 - 86 September 30

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Professor Jonathan Awerbuch
Departmnet of Mechanical Engineering and Mechanics
Drexel University
Philadelphia, Pennsylvania 19104
(215) 895-2352

Objective: To evaluate the applicability of acoustic emission monitoring to documentation of critical damage events in graphite-epoxy laminates subjected to fatigue loading.

DESIGN METHODOLOGY AND LIFE ANALYSIS OF POSTBUCKLED METAL AND COMPOSITE PANELS
Contract F33615-81-C-3208 JON: 24010154
80 June 23 - 84 August 30

Project Engineer: Lt. Mark Sobota
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: Dr. Ben Agarwal
Northrop Corporation/Aircraft Division
Structural Mechanics Research
One Northrop Avenue
Hawthorne, California 90250
(213) 970-5075

Objective: Develop analytical techniques and design procedures for metal and composite aricraft structures operating in the postbuckled range.

DAMAGE ACCUMULATION IN COMPOSITES
Contract F33615-81-C-3226 JON: 24010157
80 August 18 - 84 December 30

Project Engineer: Dr. George P. Sendeckyj
Air Force Wright Aeronautical Laboratories
AFWAL/FIBEC
Wright-Patterson AFB, Ohio 45433
(513) 255-6104 Autovon 785-6104

Principal Investigator: David A. Ulman
Structures & Design Department
General Dynamics Corporation
P. O. Box 748
Fort Worth, Texas 76101
(817) 777-3760

Objective: Develop a state-of-damage based procedure for predicting the life of composite structures subjected to spectrum fatigue loading.

Principal Investigator: Dr. David Bushnell
Lockheed Palo Alto Research Laboratory
Bldg 255
3251 Hanover
Palo Alto, California 94304
(415) 858-4027

Objective: To experimentally invstigate the behavior of optimized composite stiffened panels.

BOLTED JOINTS IN COMPOSITE STRUCTURES: DESIGN, ANALYSIS AND VERIFICATION
Contract F33615-82-C-3217 JON: 24010255
82 September 15 - 86 July 31

Project Engineer: Dr. V. B. Venkayya
Air Force Wright Aeronautical Laboratories
AFWAL/FIBRA
Wright-Patterson AFB, Ohio 45433
(513) 255-6992 Autovon 785-6992

Principal Investigator: Dr. Ramkumar
Northrop Corporation/Aircraft Division
One Northrop Avenue
Hawthorne, California 90250
(213) 970-5075

Objective: To develop reliable analytical and experimental methods for strength and life prediction of multi-fastener joints in full-scale composite structures. The end products will be a new or revised computer program for analysis and a design guide for representative aircraft joints.

SURVIVABLE COMPOSITE FUEL TANK STRUCTURES
Contract F33615-82-C-3212 JON: 24010357
82 August 2 - 85 November 30

Project Engineer: D. Oetting
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCA
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-5864

Principal Investigator: Dr. M. J. Jacobson
Northrop Corporation
Aircraft Services Division
One Northrop Avenue
Hawthorne, California 90250
(213) 970-2000

Objective: Develop guidelines for the design of composite integral fuel tanks capable of surviving hostile environments created by non-detonating projectiles and warhead fragments.

SURVIVABILITY CHARACTERISTICS OF COMPOSITE COMPRESSION STRUCTURE
Contract F33615-83-C-3228 JON: 24010365
84 March 5 - 88 May 7

Project Engineer: D. Oetting
Air Force Wright Aeronautical Laboratories
AFWAL/FIBCA
Wright-Patterson AFB, Ohio 45433
(513) 255-5864 Autovon 785-5864

Principal Investigator: Jack Avery
Boeing Military Airplane Company
Seattle, Washington
(206) 655-4373

Objective: To provide survivability/vulnerability data and analysis techniques for the design of those composite structural members of combat aircraft which are loaded primarily in compression and to provide new structural concepts

for composite compression components which are more survivable than concepts commonly used today.

COMPOSITE WING/FUSELAGE PROGRAM
Contract F33615-79-C-3203 JON: 69CW0152
79 July 1 - 87 January 2

Project Engineer: James L. Mullineaux
Air Force Wright Aeronautical Laboratories
AFWAL/FIBAC
Wright-Patterson AFB, Ohio 45433
(513) 255-6639 Autovon 785-6639

Principal Investigator: Gordon Ritchie, Program Manager
Northrop Corporatioon/Aircraft Division
One Northrop Avenue
Hawthorne, California 90250
(213) 970-5111

Objective: To develop structural design technology and durability qualification methodology for application of advanced composites to wing and fuselage primary structures of Mach 2 class fighter aircraft. Secondary efforts within the program will verify low cost fabrication methods, develop quality assurance techniques and evaluate the effects of defectys in composites.

DAMAGE TOLERANCE OF COMPOSITES
Contract F33615-82-C-3213 JON: 69CW0160
82 September 27 - 86 September 30

Project Engineer: Dr. Edvins Demuts
Air Force Wright Aeronautical Laboratories
AFWAL/FIBAC
Wright-Patterson AFB, Ohio 45433
(513) 255-6639 Autovon 785-6639

Principal Investigator: John E. McCarty, Program Manager
The Boeing Company, MS 44-56
P. O. Box 3707
Seattle, Washington 98124
(206) 655-3479

Objective: Develop comprehensive damage tolerance requirements for the design, qualification/certification, maintenance and repair of composite safety-of-flight structures; validate the requirements by designing, development testing, fabricating and testing four wing box components according to the requirements.

AIR FORCE OFFICE OF SCIENTIFIC RESEARCH

INHOUSE

NONE

GRANTS AND CONTRACTS

DAMAGE MODELS FOR CONTINUOUS FIBER COMPOSITES
84 February 01 - 85 January 31

Principal Investigator: Dr David H Allen
Department of Aerospace Engineering
Texas A&M University
College Station, TX 77843
(409) 845-7541

Program Manager: Maj David A Glasgow
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To develop a damage model for predicting strength and stiffness of continuous fiber composite structure subjected to fatigue loading, and to verify this model with experimental results.

DELAMINATION AND TRANSVERSE FRACTURE IN GRAPHITE/EPOXY COMPOSITES
84 February 01 - 85 January 31

Principal Investigator: Dr Walter L Bradley
Department of Mechanical Engineering
Texas A&M University
College Station, TX 77843
(409) 845-1259

Program Manager: Maj David A Glasgow
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To better define the deformation and fracture physics of delamination and transverse fracture in graphite epoxy composites, and to incorporate more realistic macroscopic measures of the fracture process into linear and nonlinear materials characterization.

DYNAMICS AND AEROELASTICITY OF COMPOSITE STRUCTURES
84 May 01 - 85 April 30

Principal Investigator: Dr John Dugundji
Department of Aeronautics & Astronautics
Massachusetts Institute of Technology
Cambridge, MA 02139
(617) 253-3758

Program Manager: Dr Anthony K Amos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To pursue combined experimental and theoretical investigations of aeroelastic tailoring effects on flutter and divergence of aircraft wings.

ANALYTICAL AND EXPERIMENTAL CHARACTERIZATION OF DAMAGE PROCESSES IN COMPOSITE LAMINATES
84 September 30 - 85 September 29

Principal Investigator: Dr George J Dvorak
Department of Civil Engineering
Rensselaer Polytechnic Institute
Troy, NY 12181
(518) 266-6943

Program Manager: Maj David A Glasgow
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To develop distributed damage analysis applicable to high matrix crack densities, examine damage propagation across and along ply interfaces, model damage growth from intensely damaged regions, and analyze stability and compressive strength of laminated plates containing distributed and/or concentrated damage.

INTERNAL DAMPING IN FIBER REINFORCED COMPOSITES
83 Jun 01 - 85 May 31

Principal Investigator: Dr Ronald F Gibson
Department of Mechanical Engineering
University of Idaho
Moscow, ID 83843
(208) 885-7432

Program Manager: Dr Donald R Ulrich
AFOSR/NC
Bolling AFB DC 20332-6448
(202) 767-4963

Objective: To establish quantitative relationships between the observed mechanical damping properties of organic matrix composite materials and controllable structural characteristics.

BEHAVIOR OF FIBRE REINFORCED COMPOSITES UNDER DYNAMIC TENSION
84 March 15 - 85 March 14

Principal Investigators: Dr John Harding
Dr C Ruiz
Department of Engineering Science
University of Oxford
Oxford, OX1 3PJ England

Program Manager: Dr Anthony K Amos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To characterize the mechanical behavior and failure mechanisms of carbon/epoxy, Kevlar/epoxy, and hybrid composites under tensile impact loading using specially designed split Hopkinson bar equipment.

ANALYSIS OF FATIGUE DAMAGE AND FAILURE IN COMPOSITE MATERIALS
84 September 30 - 85 September 29

Principal Investigator: Dr Zvi Hashin
Dept of Materials Science and Engineering
University of Pennsylvania
Philadelphia, PA 19104
(215) 898-8337

Program Manager: Maj David A Glasgow
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To evaluate stiffness change in laminates due to distribution of intralaminar and interlaminar cracks by the use of variational methods, and to determine the relationship between the stiffness deterioration and the strength deterioration of cracked laminates.

RESISTANCE CURVE APPROACH TO PREDICTING RESIDUAL STRENGTH OF COMPOSITES
84 August 01 - 85 July 31

Principal Investigator: Dr H P Kan
Northrop Corporation
One Northrop Avenue
Hawthorne, CA 90250
(213) 970-2134

Program Manager: Maj David A Glasgow
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To experimentally determine the Mode II delamination growth resistance of composite laminates and to develop analytical techniques for application of the R-curve concept to residual strength prediction of composite laminates with delaminations.

ULTRASONIC NDE OF DAMAGE IN CONTINUOUS FIBER COMPOSITES
84 February 01 - 85 January 31

Principal Investigator: Dr Vikram K Kinra
Department of Aerospace Engineering
Texas A&M University
College Station, TX 77843
(409) 845-1667

Program Manager: Maj David A Glasgow
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To develop, test, and implement ultrasonic nondestructive evaluation techniques to characterize damage states produced in continuous fiber composites by monotonic and fatigue loading.

FRACTURE AND LONGEVITY OF COMPOSITE STRUCTURES
82 January 01 - 85 March 14

Principal Investigator: Dr James W Mar
Department of Aeronautics & Astronautics
Massachusetts Institute of Technology
Cambridge, MA 02139
(617) 253-2426

Program Manager: Dr Anthony K Amos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To develop theoretical and semi-empirical fracture laws and failure criteria and to correlate them with extensive experimental data generated in the program.

NONLINEAR DYNAMIC RESPONSE OF COMPOSITE ROTOR BLADES
82 September 01 - 85 August 31

Principal Investigators: Dr Ozden Ochoa
Department of Mechanical Engineering
Texas A&M University
College Station, TX 77843
(409) 845-2022

Dr John J Engblom
Department of Mechanical Engineering
Texas A&M University
College Station, TX 77843
(409) 845-2813

Program Manager: Dr Anthony K Amos
AFOSR/NA
Bolling AFB DC 20332
(202) 767-4937

Objective: To develop nonlinear finite element models suitable for predicting the structural dynamic response and resulting damage of composite rotor blades under impact and other transient excitations.

INTERLAMINAR FRACTURE TOUGHNESS IN RESIN MATRIX COMPOSITES
83 January 01 - 85 February 14

Principal Investigator: Dr Lawrence W Rehfield
School of Aerospace Engineering
Georgia Institute of Technology
Atlanta, GA 30332
(404) 894-3067

Program Manager: Maj David A Glasgow
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To develop a Mode II interlaminar fracture coupon and test that can be used in both tension and compression testing, can be analyzed conveniently so that behavior can be readily interpreted and provides an experimental means for isolating Mode II contributions to fracture.

DAMAGE MODELS FOR DELAMINATION AND TRANSVERSE FRACTURE IN FIBROUS COMPOSITES
84 February 15 - 85 February 14

Principal Investigator: Dr Richard A Schapery
Department of Civil Engineering
Texas A&M University
College Station, TX 77843
(409) 845-7512

Program Manager: Maj David A Glasgow
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To develop and verify mathematical models of delamination and transverse fracture which account for local (crack tip) and global damage distributions, separating the lay-up dependent fracture energy associated with microcracking from the intrinsic fracture energy of the separation process which occur at the tip of an advancing delamination crack.

EFFECT OF LOCAL MATERIAL IMPERFECTIONS ON BUCKLING OF COMPOSITE STRUCTURAL ELEMENTS
83 June 30 - 85 August 31

Principal Investigator: Dr George J Simitses
Dept of Engineering Science and Mechanics
Georgia Institute of Technology
Atlanta, GA 30332
(404) 894-2770

Program Manager: Dr Anthony K Amos
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To investigate the effects of localized material, geometric, and process imperfections on the buckling characteristics of composite structural elements, and to incorporate them in analytical prediction methods.

COMPREHENSIVE STUDY ON DAMAGE TOLERANCE PROPERTIES OF NOTCHED COMPOSITE LAMINATES
84 September 30 - 85 September 29

Principal Investigator: Dr Albert S D Wang
Dept of Mechanical Engineering and Mechanics
Drexel University
Philadelphia, PA 19104
(215) 895-2297

Program Manager: Maj David A Glasgow
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To conduct a comprehensive analysis of the stress fields in notched laminates so as to develop a fundamental understanding of the damage mechanisms near the notch region.

RESIDUAL STRESS INDUCED DAMAGE IN COMPOSITE MATERIALS
84 February 01 - 85 January 31

Principal Investigator: Dr Y Weitsman
Department of Civil Engineering
Texas A&M University
College Station, TX 77843
(409) 845-7512

Program Manager: Maj David A Glasgow
AFOSR/NA
Bolling AFB DC 20332-6448
(202) 767-4937

Objective: To develop and verify methods for predicting damage formation, growth, and arrest due to residual stresses in fiber-reinforced, resin matrix composites.

NASA LANGLEY RESEARCH CENTER

INHOUSE

EFFECT OF FOIL TOUGHENING ON IMPACT RESISTANCE OF LAMINATES
81 May 1 - 85 May 1

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Objective: To determine the effect on impact resistance of partial interlaminar separations between layers of a laminate. Perforated mylar foil produces the partial separations.

ELASTOPLASTIC ANALYSIS OF DELAMINATION
84 October 1 - 85 September 30

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Objective: To develop a simple method to separate fracture energy into strain energy release rate and plasticity components.

MECHANICS OF LOW-VELOCITY IMPACT
81 June 1 - 85 September 30

Project Engineer: Gretchen Bostaph Murri
Mail Stop 188E
Structures Laboratory, USARTL (AVSCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Objective: From quasi-static deformation analysis, determine the influence of matrix, and fiber properties on low-velocity impact damage; develop fracture mechanics analyses for delamination growth and membrane failure; determine effect of toughened matrices on delamination; investigate failure mechanics of honey-comb supported plates.

NONLINEAR ACOUSTIC ANALYSIS OF COMPOSITES
83 July 1 - 85 June 30

Project Engineer: Dr. William P. Winfree
Mail Stop 499
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3036 FTS 928-3036

Objective: The objective of this research is to develop acoustic sensors for monitoring resin moduli and viscosity during composite cure. The sensor outputs will be used as real time inputs for controlling the cure process to obtain composite structures with maximum integrity.

SYNTHESIS OF TOUGHENED MATRIX RESIN SYSTEMS
81 October 1 - 85 September 30

Project Engineers: Dr. Terry L. St. Clair
Dr. Vernon L. Bell, Jr.
Paul M. Hergenrother
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Objective: New polymer compositions are being synthesized for evaluation as new toughened graphite composite matrix materials. Linear, thermoplastic polyimides, lightly crosslinked polysulfones, and polyesters, as well as semicrystalline polyesters, are being investigated.

TOUGHNESS TEST METHODOLOGY
80 October 1 - 85 September 30

Project Engineer: Dr. Norman J. Johnston/Dr. Jeffrey Hinkley
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Objective: To investigate, develop (if necessary), and select appropriate test methods for screening the impact resistance and fracture toughness properties of neat polymers and composites. Methodology will help guide programs to synthesize new toughened matrix resins. Edge-delamination, double-cantilever-beam, and compact-tension tests are being emphasized using a variety of tough and brittle matrix resins.

FRACTURE OF LAMINATED COUPONS
78 October 1 - 85 September 30

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Objective: To develop a methodology to predict residual strengths of damaged composite laminates using, as starting points, lamina properties or possibly the properties of the fibers and matrix. To determine the parameters that lead to tough composites.

DAMAGE TOLERANT COMPOSITE STRUCTURES
74 June 1 - 85 September 30

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Objective: To measure the ability of buffer strips and bonded stringers to increase the residual tension strength of damaged panels, and to develop an analysis to predict residual strength in terms of panel configuration and damage size.

EFFECT OF ELEVATED TEMPERATURE ON LARGE GRAPHITE/POLYIMIDE BUFFER STRIP PANELS
81 February 11 - 85 September 30
Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Objective: To experimentally determine the effect of elevated temperature on the fracture behavior of large graphite/polyimide buffer strip panels with various size buffer strips.

EFFECT OF IMPACT ON FWC FOR SPACE SHUTTLE'S SRBs
83 August - 85 September

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton Virginia 23665
(804) 865-2338 FTS 928-2338

Objective: To determine the strength loss of FWC's due to low velocity impact.

EFFECT OF MOISTURE AND ELEVATED TEMPERATURE ON GRAPHITE/EPOXY BUFFER STRIP PANELS
80 November 1 - 85 September 30

Project Engineer: C. A. Bigelow
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3047 FTS 928-3047

Objective: To experimentally determine the effect of moisture and elevated temperature on the fatigue life of graphite/epoxy buffer strip panels.

EFFECT OF DEBOND GROWTH ON CRACK PROPAGATION IN COMPOSITE PLATES REINFORCED WITH ADHESIVELY BONDED COMPOSITE STRINGERS
81 January 1 - 85 September 30

Project Engineer: C. A. Bigelow
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3047 FTS 928-3047

Objective: To analytically examine the effects of partial debonding on the crack propagation in adhesively bonded composite structures using complex variable elasticity and including a nonlinear adhesive.

WOVEN COMPOSITE BUFFER STRIP PANELS
81 January 1 - 85 September 30

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Objective: To demonstrate that buffer strip panels built with woven cloth have the crack-arresting capability of panels built with conventional prepreg tape. Damaged panels will be tested in shear and tension.

STRESS ANALYSIS OF LAMINATES WITH BEARING BYPASS LOADS
83 October 1 - 85 September 30

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Objective: To calculate stresses near loaded holes in finite-length laminates with tension-reacted and compression-reacted bearing.

ADHESIVE DEBOND CHARACTERIZATION
76 October 1 - 85 September 30

Project Engineers: Dr. W. Steven Johnson
Richard A. Everett, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 928-2715

Objective: To verify that identical specimens manufactured at different facilities using the same adhesive/adherent (7075 Al/FM-73) bonding techniques behave in a similar manner when subjected to cyclic loading. To develop an approach to calculate cyclic debond threshold and rate such that the cyclic behavior of the bondline can be predicted for any geometry (using finite elements) for a given adhesive/adherent system. To expand from metal-to-metal to composite-to-composite bonds and to examine temperature, moisture, and spectrum loading effects.

STRESS ANALYSIS OF ADHESIVE BONDS
80 October 1 - 85 September 30

Project Engineers: Richard A. Everett, Jr., USARTL (AVSCOM)
John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Objective: To review currently available finite-element routines and their applicability to the adhesive bondline stress analysis. An existing nonlinear geometric finite-element program has been modified to incorporate nonlinear material behavior and to calculate G_I and G_{II} at the debond front.

FAILURE MODES OF ADHESIVELY BONDED COMPOSITE JOINTS AND INTERLAMINAR TOUGHNESS
81 June 1 - 85 September 30

Project Engineers: Dr. W. Steven Johnson
Dr. P. D. Mangalgiri
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Objective: To conduct experimental tests to determine the failure modes and mechanisms of adhesively bonded composite joints. Adherend thickness effects for symmetric and unsymmetric double cantilever beam specimens will be examined. Further, the interlaminar fracture toughness of several matrix materials will be studied using the cracked lap shear specimen.

FATIGUE AND FRACTURE OF METAL MATRIX COMPOSITES
79 December 18 - 85 October 1

Project Engineer: Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Objective: To characterize the fatigue behavior of continuous fiber metal matrix composites under constant amplitude and spectrum loading. B/Al and SCS_2/Al are being investigated. The fracture process of the composites containing crack-like slits will be investigated.

REALISTIC ADHESIVELY BONDED JOINT ELEMENT
81 October 1 - 85 September 30

Project Engineer: Richard A. Everett, Jr.
Mail Stop 188E
Structures Laboratory, USARTL (AVSCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Objective: Three variations of a simple adhesively bonded wing splice joint have been manufactured under contract (metal-to-composite specimens) to determine the fatigue and static failure modes for a "realistic" aircraft adhesively bonded structure. The first series of tests have shown that the failure mode for both fatigue and static loading is composite delamination. No failure occurred in the adhesive bondline. Static strengths of the joints were almost equal to the strain-to-failure of the fibers.

ADHESIVE BOND CHARACTERIZATION
82 October 1 - 85 September 30

Project Engineer: Carl E. Rucker
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804)865-3047 FTS 928-3047

Objective: To measure adhesive mechanical properties in the bonded condition. Develop techniques and assess reliability. NDI will be used to investigate complex modulus and classify relative strength characteristics.

PREDICTION OF INSTABILITY-RELATED DELAMINATION GROWTH
79 January 2 - 85 September 30

Project Engineer: John D. Whitcomb
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3011 FTS 928-3011

Objective: To predict rate of instability-related delamination growth. Approximate stress analyses will be developed based on understanding gained from rigorous analyses. Experiments will be performed to obtain a data base for use by the analysis in making predictions and for verifying and improving the analysis.

PREDICTION OF STIFFNESS LOSS, RESIDUAL STRENGTH, AND FATIGUE LIFE OF UNNOTCHED LAMINATES
80 June 1 - 85 October 30

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
Structures Laboratory, USARTL (AVSCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Objective: To predict the stiffness loss, residual strength, and fatigue life of realistic unnotched laminates using baseline data from simple laminates.

DETERMINATION OF EFFECT OF RESIN TOUGHNESS ON MECHANICS OF COMPRESSION FAILURE
83 May 1 - 85 September 30
Project Engineers: John D. Whitcomb
Dr. Norman J. Johnston
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3011 FTS 928-3011

Objective: To identify parameters related to the mechanics of compression failure. To develop an analytical model to predict compression failure.

DEVELOPMENT OF INTERLAMINAR FRACTURE TOUGHNESS TESTS
84 September 30 - 85 September 30

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
Structures Laboratory, USARTL (AVSCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Objective: To develop analyses and experimental techniques to measure interlaminar fracture toughness over a wide range of mixed mode loadings for materials with brittle or tough matrix resins.

FLIGHT SERVICE EVALUATION OF COMPOSITE COMPONENTS ON COMMERCIAL AND MILITARY AIRCRAFT
72 March 1 - 90 December 31

Project Engineer: H. Benson Dexter
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2848 FTS 928-2848

Objective: To evaluate the long-term durability of composite components installed on commercial and military transport and helicopter aircraft. Over 300 components constructed of boron, graphite, and Kevlar composites will be evaluated after extended service. Components include graphite/epoxy rudders, spoilers, tail rotors, vertical stabilizers, Kevlar/epoxy fairings, doors and ramp skins, and boron/aluminum aft pylon skins. Note: Over 3.2 million total component flight hours have been accumulated since initiation of flight service in 1972. Composite components on L-1011, B-737, and DC-10 aircraft have accumulated over 28,000 flight hours each. Excellent in-service performance and maintenance experience have been achieved with the composite components.

POSTBUCKLING RESPONSE OF COMPOSITE MATERIAL SUBJECTED TO SHEAR LOADING
79 July 1 - 85 June 30

Project Engineer: Gary L. Farley
Mail Stop 188A
Structures Laboratory, USARTL (AVSCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2850 FTS 928-2850

Objective: To determine the postbuckling strength of Kevlar and Kevlar-graphite/epoxy composites under static shear and spectrum fatigue loading, as well as low-velocity and ballistic impact. This study will establish a basis for demonstrating the use of thin composite laminates beyond the point of initial shear instability. A shear fixture has been developed that virtually eliminates the adverse stresses in the corners of the shear panel.

THE ENERGY ABSORPTION OF COMPOSITE CRASHWORTHY STRUCTURE
80 August 1 - 85 December 31

Project Engineer: Gary L. Farley
Mail Stop 188A
Structures Laboratory, USARTL (AVSCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2850 FTS 928-2850

Objective: To determine the energy absorption characteristics of glass, Kevlar, and graphite/epoxy composites and to develop the analytical capability to predict the energy absorption characteristics of new composite materials. Tube specimens are being subjected to static and dynamic crushing tests. The research is focused on development of the capability to design efficient crashworthy composite structures for rotorcraft.

ADVANCED CONCEPTS FOR COMPOSITE HELICOPTER FUSELAGE STRUCTURES
83 April 1 - 85 December 31

Project Engineer: Donald J. Baker
Mail Stop 188B
Structures Laboratory, USARTL (AVSCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2850 FTS 928-2850

Objective: Investigate new design concepts for composite materials on lightly loaded helicopter fuselage structures. Trade studies will be performed using the computer code PASCO. Initial studies will be for compression loading. After testing some of the designs for panel compression, trade studies will be performed for combined compression/shear-loaded panels.

EFFECTS OF THERMAL CYCLING ON DIMENSIONAL STABILITY OF GRAPHITE/EPOXY COMPOSITES
81 October 1 - 86 September 30

Project Engineer: Dr. Stephen S. Tompkins
Mail Stop 191
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-4558 FTS 928-4558

Objective: To determine the effects of thermal cycling from 117K to 400K on dimensional stability of graphite/epoxy composites.

DIMENSIONAL STABILITY OF METAL-MATRIX COMPOSITES IN THE SPACE ENVIRONMENT
82 October 1 - 85 September 30

Project Engineer: Dr. Stephen S. Tompkins
Mail Stop 191
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-4558 FTS 928-4558

Objective: To determine and predict the dimensional changes induced by long-time exposure to the space environment and thermal cycling.

RADIATION EFFECTS ON MATERIALS FOR STRUCTURAL COMPOSITES
79 July 1 - 86 June 30

Project Engineer: Dr. Edward R. Long, Jr.
Mail Stop 399
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3892 FTS 928-3892

Objective: To determine and correlate the effects of particulate radiation exposure on the properties and chemical structure of materials for structural composites and to develop procedures for accelerated laboratory simulation of long-term missions in a space radiation environment.

EFFECT OF MICROCRACKING ON THE DIMENSIONAL STABILITY OF COMPOSITES
80 October 1 - 85 September 30

Project Engineer: David E. Bowles
Mail Stop 191
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-4558 FTS 928-4558

Objective: To develop analytical methods to predict the effect of microcracking on the dimensional stability of graphite/resin composites and correlate with experimental data.

POSTBUCKLING AND CRIPPLING OF COMPRESSION-LOADED COMPOSITE STRUCTURAL COMPONENTS
79 March 1 - 85 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2552 FTS 928-2552

Objective: To study the postbuckling and crippling of compression-loaded composite components and to determine the limitations of postbuckling design concepts in structural applications.

DESIGN TECHNOLOGY FOR STIFFENED CURVED COMPOSITE PANELS
79 October 1 - 85 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2552 FTS 928-2552

Objective: To develop verified design technology for generic advanced-composite stiffened curved panels.

COMPRESSION STRENGTH OF COMPOSITE LAMINATES WITH CUTOUTS
77 October 1 - 85 September 30

Project Engineer: Mark Shuart
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2813 FTS 928-2813

Objective: To study the effects of cutouts on the compression strength of composite structural components and to identify the failure modes that govern the behavior of compression-loaded components with cutouts.

POSTBUCKLING OF FLAT STIFFENED GRAPHITE/EPOXY SHEAR WEBS
81 July 1 - 85 September 30

Project Engineer: Marshall Rouse
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-4585 FTS 928-4585

Objective: To study the postbuckling response and failure characteristics of flat stiffened graphite/epoxy shear webs.

CURVED GRAPHITE/EPOXY PANELS SUBJECTED TO INTERNAL PRESSURE
80 October 1 - 85 September 30

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2552 FTS 928-2552

Objective: To study the effects of internal pressure on the nonlinear response and failure characteristics of stiffened graphite/epoxy panels.

POSTBUCKLING ANALYSIS OF GRAPHITE/EPOXY LAMINATES
80 October 1 - 85 September 30

Project Engineer: Dr. Manuel Stein
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2813 FTS 928-2813

Objective: To develop accurate analyses for the postbuckling response of graphite/epoxy laminates and to determine the parameters that govern postbuckling behavior.

STRUCTURAL PANEL ANALYSIS AND SIZING CODE FOR STIFFENED PANELS
79 October 1 - 85 September 30

Project Engineers: Dr. Melvin S. Anderson
Dr. W. Jefferson Stroud
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3054 FTS 928-3054

Objective: To develop an accurate analysis and structural optimization capability for stiffened composite panels subjected to inplane tension, compression, shear, normal pressure, and thermal loads.

CRASH CHARACTERISTICS OF COMPOSITE FUSELAGE STRUCTURE
82 July 1 - 85 September 30

Project Engineer: Huey D. Carden
Mail Stop 495
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3795 FTS 928-3795

Objective: To study the crash characteristics of composite transport fuselage structural components.

DAMAGE TOLERANT DESIGN TECHNOLOGY FOR COMPRESSION-LOADED COMPOSITE STRUCTURAL COMPONENTS
78 October 1 - 85 September 30

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-4052 FTS 928-4052

Objective: To develop structurally efficient design concepts for containing and resisting damage in compression-loaded composite structural components.

COMPRESSION STRENGTH OF COMPOSITE LAMINATES WITH DAMAGE AND LOCAL DISCONTINUITIES
76 October 1 - 85 September 30

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-4052 FTS 928-4052

Objective: To study the effects of impact damage and local discontinuities on the compression strength of composite structural components, to identify the failure modes that govern the behavior of compression-loaded components subjected to low-velocity impact damage, and to analytically predict failure and structural response.

CONTRACTS

IMPACT CONTACT STRESS ANALYSIS
NAG-1-222
81 November 1 - 84 December 31

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Principal Investigator: Dr. C. T. Sun
School of Aeronautics and Astronautics
Purdue University
West Lafayette, Indiana 47907
(317) 494-5130

Objective: To integrate the contact behavior and dynamic structural response to solve impact problems involving laminates under initial stress. With the aid of the previously-developed contact law, the dynamic response of the laminate will be modeled by finite elements. Impact damage will be investigated experimentally and correlated with the results of the analysis.

FRACTURE BEHAVIOR OF THICK LAMINATES
81 October 1 - 85 September 30

NAG1-

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Principal Investigator: Dr. Donald H. Morris
ESM Dept.
Virginia Polytechnic Institute and State University
Blacksburg, VA 24061

Objective: To identify potential fracture problems associated with scale-up of graphite/epoxy laminates to thicknesses of about 100 plies and to compare fracture toughness obtained from tests on center-crack, compact-tension, and bending specimens. Both through and part-through thickness slits will be considered.

THE EFFECT OF LOW VELOCITY IMPACT ON THE STRENGTH CHARACTERISTICS OF COMPOSITE MATERIALS
NAG-1-158
80 June 1 - 85 September 30

Project Engineer: Gretchen Bostaph Murri
Mail Stop 188E
Structures Laboratory, USARTL (AVSCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Principal Investigator: Dr. Thomas Moyer
School of Engineering and Applied Science
The George Washington University
Washington, DC 20052
(202) 676-6080

Objective: To develop improved analytical techniques for the full dynamic analysis of impact events on composite plates.

EXPERIMENTAL STUDIES OF IMPACT DAMAGE IN COMPOSITE LAMINATES
NAG-1-366
83 May 17 - 84 December 31

Project Engineer: Walter Illg
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Principal Investigator: Dr. I. M. Daniel
Department of Mechanical Engineering
Illinois Institute of Technology
Chicago, Illinois 60616
(312) 567-3185

Objective: To characterize impact damage in graphite/epoxy composite laminates and correlate it with transient strain and deformation history during impact. Plate and beam specimens containing embedded strain gages will be impacted with projectiles of various radii at two velocities.

STRENGTH AND TOUGHNESS OF BEARING-LOADED LAMINATES
82 October 1 - 85 September 30

NAS1-17099-28

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Principal Investigator: Dr. R. Prabhakaran
Department of Mechanical Engineering and Mechanics
Old Dominion University
Norfolk, VA 23508

Objective: To develop the basic understanding of the failure micromechanics that govern damage onset, strength, and fracture toughness of laminates subjected to combined bearing-bypass loads.

MICROMECHANICS ANALYSIS OF DELAMINATION
84 March 1 - 84 December 31

NAG-1-454

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3048 FTS 928-3048

Principal Investigator: D. E. Walrath
Composite Materials Research Group
University of Wyoming
Laramie, WY 82071

Objective: To explore correlations between calculated strain energy release rates and resin constitutive relationships.

QUANTITATIVE RECONSTRUCTIVE ULTRASONIC AND THERMAL IMAGING
NCC1-50
83 January 1 - 85 December 31

Project Engineer: Dr. Joseph S. Heyman
Mail Stop 499
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3418 FTS 928-3418

Principal Investigator: Dr. Chris Welch
Virginia Associated Research Campus
College of William and Mary
12070 Jefferson Avenue
Newport News, Virginia 23606
(804) 877-9231

Objective: To develop a state-of-the-art ultrasonic and thermal diffusivity reconstructive imaging system for quantitative materials characterization.

QUANTITATIVE PHYSICAL ANALYSIS OF IMPACT DAMAGE
NSG-1601
80 March 1 - 85 February 29

Project Engineer: Dr. Joseph S. Heyman
Mail Stop 499
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3418 FTS 928-3418

Principal Investigator: Professor James G. Miller
Laboratory for Ultrasonics
Physics Department
Washington University
St. Louis, Missouri 63130
(314) 889-6229

Objective: To improve nondestructive acoustic/ultrasonic techniques for quantitative characterization of defects in composite materials and to investigate new quantitative measurement phenomena applicable to graphite/epoxy.

NEAT RESIN-COMPOSITE PROPERTY RELATIONSHIPS
NAG-1-277
82 May 5 - 85 May 3

Project Engineer: Dr. Norman J. Johnston
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Principal Investigator: Dr. Donald F. Adams
Department of Mechanical Engineering
University of Wyoming
Laramie, Wyoming 82071
(307) 766-2371

Objective: A detailed evaluation of candidate toughened neat resin systems is being conducted, including determination of tensile modulus, tensile strength, Poisson's ratio, shear modulus, shear strength, coefficient of thermal expansion, coefficient of moisture expansion, and strain-energy-release rates. These data will be used along with appropriate micromechanics models to predict expected composite response. These predictions will be compared with composite test results to determine the validity of the model and the influence of neat resin property variations on composite response.

MECHANICAL PROPERTY STUDIES IN HIGH PERFORMANCE COMPOSITES
NAG-1-253
82 January 25 - 85 August 30

Project Engineer: Dr. Norman J. Johnston
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Principal Investigator: Dr. S. S. Sternstein
Department of Materials Engineering
Rensselaer Polytechnic Institute
Troy, New York 12181
(518) 266-6499

Objective: Develop quantitative relationships between neat resin viscoelastic properties and in situ composite resin properties. Determine what effect resin viscoelasticity has on composite mechanical properties, particularly out-of-plane properties. Dynamic mechanical spectroscopic studies are being run using both the three-point and centro-symmetric deformation geometries. Creep, stress relaxation, and biaxial studies are also planned.

DOUBLE-CANTILEVER-BEAM TEST METHOD DEVELOPMENT
L-31134B/NAS1-17074
82 February 1 - 85 September 30

Project Engineer: T. Kevin O'Brien, USARTL (AVSCOM)
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Principal Investigators: Dr. Don L. Hunston
National Bureau of Standards
Polymer Division, Building 224
Washington, DC 20234
(801) 921-3318
and
Dr. W. D. Bascom
Hercules, Inc.
Aerospace Division
Bacchus Works
Magna, Utah 84044
(801) 250-5911, ext. 3379

Objective: Develop test methods for the interlaminar fracture toughness of composite materials, with particular emphasis on the double-cantilever beam. Specimen geometry (thickness, width, and taper), stacking sequence, rate of fracture, and effects of temperature and humidity will be investigated.

ANALYSIS OF DELAMINATION AND INTERLAMINAR FRACTURE TESTS FOR COMPOSITE MATERIALS
NAG-1-286
82 April 1 - 85 October 31

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
Structures Laboratory, USARTL (AVSCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Principal Investigator: Dr. Su Su Wang
Department of Theoretical and Applied Mechanics
University of Illinois
Urbana, Illinois 61801
(217) 333-1835

Objective: Determine the correct stress singularities and account for closure of crack surfaces to accurately identify the governing parameters controlling delamination in composite laminates. Particular emphasis is placed on analyzing candidate test configurations for interlaminar fracture toughness measurement.

DEVELOPMENT OF IMPACT/SOLVENT-RESISTANT THERMOPLASTIC MATRICES
NAS1-16808
81 September 4 - 85 December 31

Project Engineer: Paul M. Hergenrother
Mail Stop 226
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3041 FTS 928-3041

Principal Investigator: Chad B. Delano
Acurex Corporation
Aero-Therm Division
485 Clyde Avenue
Mountain View, California 94042
(415) 964-3200, ext. 3820

Objective: Candidate aliphatic-aromatic heterocyles are being synthesized to develop an impact-and-solvent resistant thermoplastic with acceptable processability in the 600°F range. Heteroaromatics being investigated include polyimides, N-arylenepolybenzimidazoles, and polybenzimidazoles containing both rigid and soft segments.

FRACTURE AND CRACK GROWTH IN ORTHOTROPIC LAMINATES
NSG-1606
79 July 1 - 85 September 30

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Principal Investigator: Dr. Jonathan Awerbuch
Department of Mechanical Engineering
Drexel University
Philadelphia, Pennsylvania 19104
(215) 895-2291

Objective: To explore the fracture characteristics of graphite/polyimide composites at elevated temperatures using laminates with slits.

FRACTURE AND CRACK GROWTH IN ORTHOTROPIC LAMINATES
NSG-1297
74 October 16 - 85 October 15

Project Engineer: C. C. Poe, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2338 FTS 928-2338

Principal Investigator: Dr. James G. Goree
Department of Mechanical Engineering
Clemson University
Clemson, South Carolina 29631
(803) 656-3291

Objective: To develop analyses that predict strength of buffer strip panels using models that treat the fiber and matrix as discrete elements.

THE VISCOELASTIC CHARACTERIZATION AND LIFETIME PREDICTION OF STRUCTURAL ADHESIVES
NAG-1-227
81 November 1 - 85 October 31

Project Engineer: Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Principal Investigator: Dr. H. F. Brinson and Dr. D. Post
Department of Engineering Science and Mechanics
Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061
(703) 961-6627

Objective: To develop a procedure to predict the failure of adhesive joints where service life must span 10 to 20 years using, as a basis, analytical projections or extrapolations from short-time test data. Moire fringe analysis of bonded joints to define deformations along the length and through the thickness of bondlines.

CRAZE MICROMECHANICS IN VISCOELASTIC AEROSPACE MATERIALS
NAG-1-278
82 May 21 - 85 May 20

Project Engineer: John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Principal Investigator: Professor C. C. Hsiao
Department of Aerospace Engineering and Mechanics
University of Minnesota
107 Akerman Hall
Minneapolis, MN 55455
(612) 373-2670

Objective: To study the basic mechanisms involved in craze formation in viscoelastic materials.

ANALYSIS OF A DELAMINATION FRONT
NAG-1-474
84 May 1 - 85 April 30

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Principal Investigator: Dr. W. G. Knauss
Aeronautics and Applied Mechanics Department
California Institute of Technology
Pasadena, California 91125

Objective: To compare toughening mechanisms along a delamination front extending from an edge to the interior of a specimen.

IN-SITU STUDY OF DELAMINATION TOUGHENING MECHANISMS
NAG-1-443
84 February 1 - 84 December 31

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Principal Investigator: Dr. W. L. Bradley
Department of Mechanical Engineering
Texas Engineering Experiment Station
Texas A&M University
College Station, Texas 77843

Objective: To identify the toughening (deformation) mechanisms associated with delamination in tough composites.

3-D STRESS ANALYSIS OF A DELAMINATION FRONT
NAS1-17808
84 August 11 - 85 August 10

Project Engineer: Dr. John H. Crews, Jr.
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Principal Investigator: Dr. I. S. Raju
Analytical Services and Materials
103 Winder Road
Tabb, Virginia 23602

Objective: To calculate edge effects on strain energy release rate for a Mode I delamination specimen.

FRACTURE AND FATIGUE MECHANISM OF ADHESIVELY BONDED JOINTS
NAG-1-425
83 October 1 - 85 September 30

Project Engineer: Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Principal Investigator: Dr. Shankar Mall
Department of Engineering Mechanics
University of Missouri-Rolla
Rolla, Missouri 65401
(314) 341-4599

Objective: Develop a further understanding of adhesively bonded joints by conducting debond studies at different stress ratios and developing fracture toughness data on new adhesive systems. Fracture toughness and debond growth behavior of room-temperature and high-temperature adhesives.

FATIGUE CRACK GROWTH IN ADHESIVE JOINTS UNDER MODE I-III LOADING
NAS1-17567
83 October 1 - 85 September 30

Project Engineer: Dr. W. S. Johnson
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2715 FTS 928-2715

Principal Investigator: Dr. E. J. Ripling
Materials Research Laboratory, Inc.
One Science Road
Glenwood, Illinois 60425
(312) 755-8760

Objective: To develop debond growth rate data under mixed-mode I and III loading. These data will be compared with the mode I and mixed-mode I and II data developed in-house.

FATIGUE DAMAGE IN NOTCHED COMPOSITE LAMINATES UNDER TENSION-COMPRESSION CYCLIC LOADS
NAG-1-232
82 January 1 - 85 January 1

Project Engineer: Dr. T. Kevin O'Brien
Mail Stop 188E
Structures Laboratory, USARTL (AVSCOM)
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2093 FTS 928-2093

Principal Investigator: Dr. Wayne W. Stinchcomb
Department of Engineering Science and Mechanics
Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061
(703) 961-5316

Objective: To determine life-limiting fatigue damage mechanisms in graphite/epoxy laminates containing open holes and subjected to tension-compression fatigue loading.

DELAMINATION GROWTH IN COMPOSITE LAMINATES
NAG-1-475
84 June 1 - 85 May 31

Project Engineer: John D. Whitcomb/Wolf Elber
Mail Stop 188E
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3011 FTS 928-3011

Principal Investigator: John W. Gillespie, Jr.
208 Evans Hall
University of Delaware
Newark, Delaware 19711

Objective: Investigate instability related delamination growth in a multi-parameter test series, and compare the results with several analyses.

ANALYSIS OF WOVEN FABRIC REINFORCED COMPOSITES
NAS1-17205
82 November 1 - 86 January 15

Project Engineer: H. Benson Dexter
Mail Stop 188B
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2848 FTS 928-2848

Principal Investigator: Norris Dow
Materials Sciences Corporation
Gwynedd Plaza II
Bethlehem Pike
Spring House, Pennsylvania 19477
(215) 542-8400

Objective: To develop analytical methods to understand and predict the physical behavior of woven fabric reinforced composites, extend micromechanics methods to analysis of strength and toughness properties, evaluate potential of improved fabric designs, and develop guidelines for improved weaves. Included will be two-dimensional and three-dimensional woven fabrics with potential for improved fracture toughness and impact resistance.

EFFECTS OF HIGH-ENERGY RADIATION ON THE MECHANICAL PROPERTIES OF GRAPHITE FIBER REINFORCED EPOXY RESINS
NSG-1562
79 October 1 - 86 December 31

Project Engineer: Dr. Edward R. Long, Jr.
Mail Stop 399
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-3892 FTS 928-3892

Principal Investigator: Dr. Raymond E. Fornes
Departments of Physics and Textiles
North Carolina State University
Raleigh, North Carolina 27650
(919) 737-2503/3231

Objective: To investigate the effects of high-energy radiation on graphite fiber composites by study of composite curing effects, radiation exposure rates, mechanical fracture surfaces, and electron spin resonance properties.

ENVIRONMENTAL EXPOSURE EFFECT ON COMPOSITE MATERIALS FOR COMMERCIAL AIRCRAFT
NAS1-15148
77 November 1 - 88 November 30

Project Engineer: Dr. Ronald K. Clark
Mail Stop 191
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-4557 FTS 928-4557

Principal Investigators: Martin Gibbons and Randy Coggeshall
Boeing Commercial Airplane Company
P. O. Box 3707
Seattle, Washington 98124
(206) 251-2284

Objective: To provide technology in the area of environmental effects on graphite/epoxy composite materials, including long-term performance of advanced resin-matrix composite materials in ground and flight environments.

EFFECTS OF STRESS CONCENTRATIONS IN COMPOSITE STRUCTURES
NSG-1483
78 January 15 - 85 January 14

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2552 FTS 928-2552

Principal Investigators: Dr. Wolfgang G. Knauss
Dr. Charles D. Babcock
California Institute of Technology
Pasadena, California 91125
(213) 356-4524/4528

Objective: To study the effects of low-speed impact damage in composite structural components using high-speed motion pictures and to develop an analytical procedure for the propagation of the resulting impact damage.

COMPOSITE LAMINATE FREE EDGE REINFORCEMENT CONCEPTS
NAG-1-389
83 September 15 - 84 December 15

Project Engineer: Mark J. Shuart
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2813 FTS 928-2813

Principal Investigator: Dr. Robert M. Jones
Department of Engineering Science and Mechanics
Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061

Objective: To analyze and test free edge reinforcement concepts for delamination-prone laminates.

ADVANCED COMPOSITE STRUCTURAL DESIGN TECHNOLOGY FOR COMMERCIAL TRANSPORT AIRCRAFT
NAS1-15949
79 September 24 - 85 September 23

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2552 FTS 928-2552

Principal Investigator: John N. Dickson
Lockheed-Georgia Company
86 South Cobb Drive
Marietta, Georgia 30063
(404) 424-3085

Objective: To design, analyze, fabricate, and test generic advanced-composite structural components for transport aircraft applications in order to develop verified design technology.

STRUCTURAL OPTIMIZATION FOR IMPROVED DAMAGE TOLERANCE
NAG-1-168
81 September 1 - 85 October 15

Project Engineer: Dr. James H. Starnes, Jr.
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-2552 FTS 928-2552

Principal Investigator: Dr. Raphael T. Haftka
Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061
(703) 961-4860

Objective: To develop a structural optimization procedure for composite wing boxes that includes the influence of damage-tolerance considerations in the design process.

COMPRESSION FAILURE MECHANISMS OF COMPOSITE STRUCTURES
NAG-1-295
82 September 1 - 85 August 31

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-4052 FTS 928-4052

Principal Investigator: Dr. H. Thomas Hahn
Washington University
Campus Box 1087
St. Louis, Missouri 63130
(314) 889-6052

Objective: To establish the effects of material properties on microbuckling and the shear crippling failure mode in order to design stronger, more damage tolerant composite structures.

DEFORMATION MEASUREMENTS OF COMPOSITE MULTI-SPAN BEAM SHEAR SPECIMENS BY MOIRE INTERFEROMETRY
NAG-1-481
83 May 1 - 84 December 31

Project Engineer: Dr. Jerry G. Williams
Mail Stop 190
NASA Langley Research Center
Hampton, Virginia 23665
(804) 865-4052 FTS 928-4052

Principal Investigator: Dr. Daniel Post
Virginia Polytechnic Institute and State University
Blacksburg, Virginia 24061
(703) 961-6651

Objective: To accurately measure the transverse deformations and strains of a short multiple-span composite beam for comparison with theoretical predictions.

NASA LEWIS RESEARCH CENTER

INHOUSE

SIMPLIFIED COMPOSITE MICROMECHANICS EQUATIONS
80 October 1 - 84 December 30

Project Engineer: Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6138

Objective: Develop composite micromechanics equations with and without interface for predicting hygrothermomechanical properties and validate predicted results with finite element analysis.

FINITE ELEMENT SUBSTRUCTURING FOR COMPOSITE MECHANICS
82 September 15 - 85 December 30

Project Engineers: John J. Caruso/Pappu L. N. Murthy
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-5366

Objective: Develop super element finite element models for describing composite micromechanics behavior and stress concentrations in angleplied laminates.

CRACKED COMPOSITE CHARACTERIZATION
82 July 7 - 85 September 30

Project Engineers: Thomas B. Irvine/Carol A. Ginty
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-5367

Objective: Conduct experimental/theoretical investigations using RUSCAN/CODSTRAN (Real-Time Ultrasonic C-Scanning/Composite Durability Structural Analysis) to characterize progressive fracture and attendant failure modes in fiber composites with and without defects and subjected to hygrothermomechanical environments.

CODSTRAN-CONTINUING DEVELOPMENT
82 July 7 - 85 December 31

Project Engineer: Thomas B. Irvine
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-5367

Objective: Continue development/documentation of CODSTRAN (Composite Durability Structural Analysis) with respect to participating fracture modes, combined stress failure criteria, complex loading conditions and corroboration with experimental data.

FAILURE MODES AND FRACTURE SURFACE CHARACTERISTICS
80 September 1 - 84 December 15

Project Engineer: Carol A. Ginty
MS 49-6
NASA Lewis Research Center

Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Objective: Identify/characterize failure modes and attendant fracture surface characteristics in angleplied laminates subjected to uniaxial and combined loads.

LIFE/DURABILITY IN HYGROTHERMOMECHANICAL ENVIRONMENTS
81 June 1 - 84 December 31

Project Engineers: Carol A. Ginty/Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Objective: Continue application and experimental corroboration of Lewis-developed hygrothermomechanical theory to different fiber composites and under various adverse loading conditions.

ICAN-INTEGRATED COMPOSITE ANALYSIS COMPUTER CODE - INTERACTIVE
82 October 15 - 85 September 14

Project Engineers: Pappu L. N. Murthy/Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Objective: Include interactive capability in the general purpose, integrated computer program (code) for fiber composite structural/stress analysis and for composite mechanics (ICAN) and initiate expert system features.

HYGROTHERMOMECHANOCHRONIC THEORY
83 March 1 - 86 April 30

Project Engineer: Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Objective: Develop a unified hygro-thermo-mechano-chronic (time) theory to predict the hygrothermomechanochronic behavior of fiber composites including damping, temperature rise due to damping and attendant degradation effects, and corroborate with experimental data.

N. L. COBSTRAN
82 January 4 - 85 September 30

Project Engineer: Dale A. Hopkins
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-5366

Objective: Extend COBSTRAN (Composite Blade Structural Analysis) to nonlinear thermoviscoplastic structural analysis for high temperature fiber composite turbine blades.

ANALYSIS OF ADVANCED TURBOPROPS
81 January 15 - 84 December 31

Project Engineer: Robert A. Aiello
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6272

Objective: Use COBSTRAN to predict the structural behavior of advanced swept turboprops made with a composite-shell and metal-spar and to conduct parametric studies for the influence of composite system and laminate configuration on structural behavior.

FAILURE MODES OF TUNGSTEN FIBER REINFORCED SUPERALLOYS
81 October 1 - 85 September 30

Project Engineer: Donald W. Petrasek
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6284

Objective: To evaluate the failure of TFRS specimens subjected to combined cyclic stress and temperature conditions as well as steady state stress and temperature conditions to develop failure models to predict performance.

IMPROVED TOUGHNESS HIGH TEMPERATURE RESINS
83 October 1 - 84 October 30

Project Engineer: Kenneth J. Bowles
MS 49-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6967

Objective: To achieve a fundamental understanding of the factors which control the toughness characteristics of high temperature polymer matrix composites and to evolve criteria for predicting composite performance.

ULTRASONIC ASSESSMENT OF SHUTTLE FILAMENT - WOUND CASE (FWC) MATERIAL
83 April 1 - 85 March 31

Project Engineer: Alex Vary
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6357
Principal Investigator: Harold Kautz

Objective: Study applications of backscatter, pulse-echo, and acousto-ultrasonic approaches to assessment of initial and post-use state of FWC material with emphasis on assessment of mechanical properties and degradation/reusability.

CERAMIC MATRIX COMPOSITES
81 September 30 - 84 September 30

Project Engineer: Dr. S. R. Levine
MS 105-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6150

Objective: To develop and evaluate processing methods for the preparation of ceramic matrix composites reinforced by continuous ceramic fibers to provide new, advanced materials for aerospace applications.

ADVANCED COMPOSITE MICROMECHANICS
81 September 30 - 84 September 30

Project Engineer: Dr. J. A. DiCarlo
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6602

Objective: To determine the principal macro- and microstructural factors which control deformation, strength and toughness of advanced composite materials.

CARBON-CARBON COMPOSITES
82 October 1 - 85 October 30

Project Engineer: Dr. S. R. Levine
MS 105-1
NASA Lewis Research Center
Cleveland, OH 44135
(216) 433-4000 FTS 294-6150

Objective: To provide fundamental understanding of c/c composites in the area of environmental protection as a foundation from which to develop advanced systems with improved oxidation resistance and mechanical behavior.

NASA LEWIS RESEARCH CENTER

CONTRACTS/GRANTS

DYNAMIC DELAMINATION
NAG 3-211
81 December 15 - 85 June 30

Project Engineer: Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Principal Investigator: C. T. Sun
School of Aeronautics and Astronautics
Purdue University
West Lafayette, Indiana 47907
(317) 494-5125

Objective: Develop analytical/experimental methods to describe and characterize dynamic interlaminar delamination propagation in fiber composites.

TEST METHODS AND CHARACTERIZATION OF HIGH TEMPERATURE COMPOSITES
NAG 3-377
82 December 10 - 86 December 9

Project Engineer: Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Principal Investigator: John F. Mandell
Department of Materials Science and Engineering
Massachusetts Institute of Technology
Cambridge, Massachusetts 02139
(617) 253-7181

Objective: Develop test methods and characterize the thermomechanical behavior of high temperature fiber composites.

ADVANCED COMPOSITE COMBUSTOR STRUCTURAL CONCEPTS
NAS 3-23284
81 April 1 - 84 December 31

Project Engineer: Robert L. Thompson
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-5366

Principal Investigator: Robert P. Lohmann
Pratt and Whitney Aircraft
400 Main Street
East Hartford, Connecticut 06108
(203) 565-7778

Objective: Conduct preliminary design and evaluation study of an advanced combustor using high temperature composite materials.

EFFECTS OF ENVIRONMENT AND DEFECTS ON HIGH STRAIN RATE PROPERTIES OF COMPOSITES
NAG 3-423
83 May 15 - 86 May 14

Project Engineer: Christos C. Chamis
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6831

Principal Investigator: Isaac M. Daniel
Mechanical Engineering Department
Illinois Institute of Technology
Chicago, Illinois 60616
(312) 567-3186

Objective: Develop experimental procedures to study the influence of environment (moisture and temperature) and defects of the high-strain-rate properties of fiber composites.

STRUCTURAL DESIGN STUDY OF LOW SPEED PROPELLERS
NAS 3-23924
83 April 22 - 84 December 30

Project Engineer: Robert A. Aiello
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6272

Principal Investigator: Bennett M. Brooks
Hamilton Standard
Windsor Locks, Connecticut 06906
(203) 623-1621, ext. 5611

Objective: Identify the most promising propeller configurations incorporating advanced concepts and materials and provide optimized designs.

STAEBL
NAS 3-22525
80 September 30 - 85 December 31

Project Engineer: Murray S. Hirschbein
MS 49-6
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6272

Principal Investigator: Kenneth W. Brown
Pratt and Whitney Aircraft
400 Main Street
East Hartford, Connecticut 06108
(203) 565-7053

Objective: Develop a formalized optimum design procedure for engine blades and advanced turboprops made using advanced structural concepts and materials and meet all the aerothermomechanical design requirements in aircraft engine environments.

ULTRASONIC STRESS WAVES CHARACTERIZATION OF COMPOSITE MATERIALS
82 October 1 - 85 September 30

Project Engineer: Alex Vary
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6357

Principal Investigators: E. G. Henneke, II
J. C. Duke,, Jr.
W. W. Stinchcomb
Engineering Science & Mechanics Department
Virginia Polytechnic Institute
Blacksburg, Virginia 24061
(703) 961-5316

Objective: Establish signal acquisition and analysis methodologies and computer algorithms for acousto-ultrasonic stress wave factor (SWI) measurement of composite materials.

INVESTIGATION OF INTERFACIAL PHASE FORMATION IN FIBER REINFORCED CERAMIC MATRIX COMPOSITE MATERIALS
83 March 1 - 84 October 30

Project Engineer: Dr. D. R. Behrendt
MS 106-1
NASA Lewis Research Center
Cleveland, Ohio 44135
(216) 433-4000 FTS 294-6602

Principal Investigator: Dr. F. E. Wawner
University of Virginia
Charlottesville, Virginia 22901

Objective: To fabricate and fully characterize low density, high performance fiber reinforced ceramic matrix composite materials with respect to structural and microstructural responses to thermal treatment.

NAVAL AIR SYSTEMS COMMAND
WASHINGTON, D.C. 20361

INHOUSE

FATIGUE OF COMPOSITES UNDER COMPLEX LOADS
79 October - 85 September

Project Engineer: Dr. P. W. Mast
Naval Research Laboratory
Washington, D.C. 20375
(202) 767-2165 Autovon 297-2165

Objective: Develop a capability for predicting the structural response and initiation of failure in composite laminates and bonded joints under complex cyclic loading.

CONTRACTS

FATIGUE LIFE AND RESIDUAL STRENGTH OF COMPOSITE STRUCTURES
83 September - 85 September

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-7443 Autovon 222-7443

Principal Investigators: Dr. J. Yang and
Dr. D. Jones
The George Washington University
Washington, D.C. 20052
(202) 676-6929

Objective: Develop statistical models to describe fatigue life and residual strength of composite structures including bolted and bonded composite joints.

STRENGTH/SEALING CHARACTERISTICS OF FASTENERS IN COMPOSITES
83 September - 85 March

Project Engineer: Dr. D. R. Mulville
Naval Air Systems Command
Washington, D.C. 20361
(202) 692-7443 Autovon 222-7443

Principal Investigator: Mr. Sam Dastin
Grumman Aerospace Corporation
Bethpage, NY 11714
(516) 575-2754

Objective: Conduct experimental studies to quantify the strength/sealing characteristics of fasteners for composite fuel containing structures.

NAVAL AIR DEVELOPMENT CENTER
AIRCRAFT AND CREW SYSTEMS TECHNOLOGY DIRECTORATE
WARMINSTER, PA 18974

INHOUSE

COMPOSITE IMPACT RESISTANCE
74 March - 85 September

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-1330 Autovon 441-1330

Objective: Ascertain the impact response of generic composite structural elements and identify the physical mechanisms associated with impact damage and the critical parameters governing impact response.

ANALYTICAL MODELING OF COMPOSITE FATIGUE
84 October - 85 September

Project Engineer: Lee W. Gause
Naval Air Development Center
ACSTD/6043
Warminster, PA 18974
(215) 441-1330 Autovon 441-1330

Objective: Develop and experimentally verify a delamination growth criterion for spectrum fatigue based upon critical strain-energy-release rate computations.

CONTRACTS

DESIGN OF HIGHLY LOADED COMPOSITE JOINTS
AND ATTACHMENTS FOR TAIL STRUCTURES
N62269-82-C-0239
82 February - 85 January

Project Engineer: Ramon Garcia
Naval Air Development Center
ACSTD/60432
Warminster, PA 18974
(215) 441-1321 Autovon 441-1321

Principal Investigator: S. W. Averill
Northrop Corporation
Aircraft Group
Hawthorne, CA 90250
(213) 970-3442

Objective: To develop composite designs which will permit the use of metal to composite bolted root attachments in aircraft tail structures as an alternative to high-load transfer adhesive bonded titanium step joints. To improve damage tolerance, survivability and repairability over current composite designs. Structural efficiency, manufacturing feasibility and quality assurance requirements will be determined.

DESIGN OF HIGHLY LOADED COMPOSITE JOINTS
AND ATTACHMENTS FOR WING STRUCTURES
N62269-82-C-0238
82 February - 85 April

Project Engineer: Ramon Garcia
Naval Air Development Center
ACSTD/60432
Warminster, PA 18974
(215) 441-1321 Autovon 441-1321

Principal Investigator: M. J. Ogonowski
McDonnell Aircraft Co.
P. O. Box 516
St. Louis, MO 63166
(314) 233-8630

Objective: To develop composite designs which will permit the use of metal to composite bolted root attachments in aircraft wing structures as an alternative to high-load transfer adhesive bonded titanium step joints. Strain concentration around fastener holes, fatigue and environmental affects, damage tolerance and repairability for each concept will be determined.

COMPOSITE DEFECT CRITICALITY
N62269-82-C-0750
82 September - 84 September

Project Engineer: Dr. William R. Scott
Naval Air Development Center
ACSTD/6063
Warminster, PA 18974
(215) 441-2412 Autovon 441-2412

Principal Investigator: Dr. S. N. Chatterjee
Materials Sciences Corporation
Gwynedd Plaza II, Bethlehem Pike
Spring House, PA 19477
(215) 542-8400

Objective: Development of a comprehensive capability for assessing the criticality of defects in graphite/epoxy structures. Emphasis will be placed on identification of damage utilizing NDT method and fracture mechanics modeling of the effects of damage to structural performance.

DEVELOPMENT OF HIGH STRAIN COMPOSITE WING
N62269-81-C-0727
81 September - 84 March

Project Engineer: Mark Libeskind
Naval Air Development Center
ACSTD/60431
Warminster, PA 18974
(215) 441-1685 Autovon 441-1685

Principal Investigator: J. Bruno
Grumman Aerospace Corporation
Bethpage, NY 11714
(516) 575-6295

Objective: Design and evaluate an advanced composite wing which operates at significantly higher strain levels than current composite wings resulting in significant weight savings. Emphasis will be placed upon damage tolerance, survivability, durability and repairability.

OFFICE OF NAVAL RESEARCH
ARLINGTON, VA 22217

CONTRACTS

FLAW GROWTH AND FRACTURE OF COMPOSITE MATERIALS
AND ADHESIVE JOINTS
N00014-79-C-0579
July 83 - June 85

Project Engineer: Dr. Yapa Rajapakse
OFFICE OF NAVAL RESEARCH
Mechanics Division, Code 432S
Arlington, VA 22217
(202) 696-4306 Autovon 226-4306

Principal Investigator: Dr. S. S. Wang
University of Illinois
Department of Theoretical and Applied Mechanics
Urbana, Illinois 61801
(217) 333-1835

Objective: Analytical and numerical studies will be conducted of flaw growth and Fracture in Fiber Composite Laminates and adhesively bonded structural joints under static and dynamic loading conditions.

DAMAGE ACCUMULATION AND RESIDUAL PROPERTIES OF COMPOSITES
N00014-82-K-0572
July 82 - November 85

Project Engineer: Dr. Yapa Rajapakse
Office of Naval Research
Mechanics Division, Code 432S
Arlington, VA 22217
(202) 696-4306 Autovon 226-4306

Principal Investigator: Prof. I.M. Daniel
Illinois Institute of Technology
Department of Mechanical Engineering
Chicago, Illinois 60616
(312) 567-3186

Objective: Investigate damage mechanisms and damage accumulation in graphite/epoxy laminates for the development of models for predicting residual stiffness, residual strength, and residual life.

INVESTIGATIONS OF ENVIRONMENTAL EFFECTS AND ENVIRONMENTAL
DAMAGE IN COMPOSITES
N00014-82-K-0562
October 84 - September 87

Project Engineer: Dr. Yapa Rajapakse
Office of Naval Research
Mechanics Division, Code 423S
Arlington, VA 22217
(202) 696-4306 Autovon 226-4306

Principal Investigator: Prof. Y. Weitsman
Texas A&M University
Department of Civil Engineering
College Station, Texas 77843
(713) 845-7512

Objective: Research will be conducted to study the effects of stress and moisture on the mechanical response of graphite/epoxy composites. Special attention will be given to environmental induced damage growth and its effect on compressive and shear response.

MONITORING ACOUSTIC EMISSION IN IMPACT
DAMAGED COMPOSITES
N00014-84-K-0460
June 84 - May 85

Project Engineer: Dr. Yapa Rajapakse
Office of Naval Research
Mechanics Division, Code 423S
Arlington, VA 22217
(202) 696-4306 Autovon 226-4306

Principal Investigator: Dr. J. Awerbuch
Drexel University
Department of Mechanical Engineering and Mechanics
Philadelphia, PA 19104
(215) 895-2291

Objective: Investigations of damage in graphite/epoxy laminates due to normal and oblique impact will be carried out using a variety of experimental techniques. In particular, the use of acoustic emission for damage assessment will be explored fully.

SUPPRESSION OF DELAMINATION IN COMPOSITE LAMINATES SUBJECTED
TO IMPACT LOADING
N00014-84-K-0554
July 84 - June 86

Project Engineer: Dr. Yapa Rajapakse
Office of Naval Research
Mechanics Division, Code 423S
Arlington, VA 22213
(202) 696-4306 Autovon 226-4306

Principal Investigator: Dr. C. T. Sun
Purdue University
West Lafayette, IN 47907
(317) 494-5130

Objective: Research will be performed to investigate and establish quantative models for delamination growth in composite laminates specifically designed to suppress delamination by the use of 3-D stitching reinforcement and soft adhesive layers.

CONSTRUCTION OF NON-LINEAR MODEL FOR BINARY METAL MATRIX COMPOSITES
N00014-84-K-0468
July 84 - June 86

Project Engineer: Dr. A. S. Kushner
Office of Naval Research
Mechanics Division, Code 423S
Arlington, VA 22213
(202) 696-4306 Autovon 226-4306

Principal Investigator: Prof. H. Murakami
University of California, San Diego
La Jolla, CA 92093
(619) 452-3821

Objective: Non-linear theory for metal matrix composites will be developed, based on variational principles and multi-variable asmptotic expansion techniques. The theory will account for the effect of fiber breakage, fiber-matrix debonding and slip, matrix plasticity and delamination.

METAL MATRIX COMPOSITE INTERFACES
N00014-84-K-0495
September 84 - August 87

Project Engineer: Dr. A.S. Kushner
Office of Naval Research
Mechanics Division, Code 423S
Arlington, VA 22213
(202) 692-4306 Autovon 226-4306

Project Engineer: Prof. A.S. Argon
MIT
Department of Mechanical Engineering
Cambridge, MA 02139
(617) 253-2217

Objective: Research will be conducted to develop the micro-mechanical model of the interface in metal matrix composites which have the features of predictability for the purpose of optimizing existing fiber-matrix systems.